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THERMODYNAMICS OF WORKING GASES IN ATOMIC ROCKETS

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In some fields of technical application, e.g., in nuclear working fluid rockets, we encounter the problem of accelerating a working fluid which has been heated in a chamber to a very high enthalpy, up to the greatest velocity possible, without exceeding the tolerable heat transfer to the walls of the chamber.

Assuming isentropic expansion and a state of complete equilibrium in the chamber and in the exhaust nozzle, we at first investigate ordinary hydrogen as a particularly interesting example of a working fluid. Its thermodynamic functions and exhaust velocities are calculated at pressures between 10^{-5} and 10^{+2} atmospheres and up to temperatures of 10,000° K. A supplementary study of heat transfer due to radiation and due to convection will be done separately. Here, instead of using absolute values which are difficult to determine, parameters are introduced for the heat transfer rates, which suffice for the purposes of comparison. These are as follows: (1) the *temperature* for the heat transfer by radiation, and (2) the so-called *convection product*, consisting of a product of the gas density, gas enthalpy and the mean molecular velocity, for heat transfer by convection.

The results of quantitative calculation, shown in diagrammatic form, indicate that—by employing diathermal working gases of low radiation power, e.g., hydrogen, and for low chamber pressures—exhaust velocities of up to approximately 30,000 m./sec. can be attained—with the same amount of heat transfer as occurs in the combustion chambers of chemical rockets already tested, with exhaust velocities ranging between 2,000 and 3,000 m./sec.

An approximate comparative investigation of helium, as prototype of light noble gases, shows—in the range of moderate enthalpies—higher exhaust velocities (than for hydrogen) for the same amount of convective heat transfer. However, at the same time there appear much higher gas temperatures than in the case of hydrogen. Therefore, with higher enthalpies the originally favourable aspects are reversed to unfavourable ones due to increased ionization and radiation. This occurs before the desired high values for the exhaust velocity of above 10,000 m./sec. are even reached.

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Nuclear physics to-day permits the realization of practically unlimited concentrations of energy per unit of mass. For instance, in 1932, at the first successful attempt to fuse lithium and hydrogen into helium, i.e., the reaction $Li^7 + H^1 = 2He^4 + 17{\cdot}3$ MeV or $6{\cdot}6 \times 10^{-16}$ k.cal. were liberated in a individual reaction. This is equal to a concentration of energy gH of $5 \times 10^{+10}$ k. cal per kg. of the mass remaining from the reacting materials after the reaction. In comparison, it may be mentioned that in the case of the recombination of two hydrogen atoms to a hydrogen molecule, this means in a typical molecular reaction of highly exothermic character, the energy concentration is only about $5 \times 10^{+4}$ k.cal. per kg. mass.

Nothing would be more desirable than the possibility of application of such high concentrations of energy in rocketry, since this would make it possible to increase one of the most important factors for any particular flight performance, i.e., the ideal exhaust velocity $c_{H_1} = \sqrt{2gH}$, to any desired degree. However, as long as we only know of rockets with combustion chambers limited by material walls, high concentrations of energy beyond a certain limit cannot be controlled any more in combustion chamber and nozzle, even assuming that the walls of the combustion chamber would remain impenetrable to the energy-carrying particles. There exist two reasons for this:

1. Because the convective heat transfer from the heated gas to the walls, as will be shown later, is always proportional to a power $n \leqslant 1$ of the product of total enthalpy, H, and the gas pressure, p,—beyond a certain enthalpy value—a further increase in enthalpy will only be possible by a simultaneous and almost equal decrease in gas pressure, even in the favourable case of a completely radiation-free gas.

2. The pressure of the heated gas itself, however, can be lowered only so far, as long as comformity with the laws of continuous media are guaranteed in the flow. In the case of molecular flow, where a protective boundary layer is lacking, the convective heat transfer to the walls surrounding the chamber is directly related to the enthalpy, and the exhaust velocities possible are then determined by the wall temperatures which set in as a result of energy equilibrium between the gas and the walls. These features appear just in the instant when the value of the mean free path Λ becomes comparable with the thickness of the flow boundary layer; they are distinctly present when Λ reaches the same order of magnitude as that of the dimensions of the combustion chamber. For monatomic hydrogen, for example, the mean free paths in metres are in the region of $\Lambda_H = 1{\cdot}3 \times 10^{-10}\ T/p$, where the pressure, p, is measured in atmospheres and the temperature, T, in degrees K. For 4,000° K. and 10^{-5} atmospheres, the mean free path becomes thus $\Lambda_H = 0{\cdot}19$ m. As soon as the homogeneous gas is going to be transformed into a plasma, the mean free paths for each component of the plasma have to be evaluated separately. With $p = 10^{-5}$ atmospheres and $T = 8{,}000°$ K., where hydrogen atoms, protons and electrons have approximately equal partial pressures, in a state of equilibrium the free paths of the individual components of the gas mixture become $\Lambda_H = 0{\cdot}12\ m$, $\Lambda_{H^+} = 4{\cdot}94\ m$ and $\Lambda_{e^-} = 6{\cdot}98\ m$. Hence, just the particles with

the highest recombination probability first cease to participate in the formation of the boundary layer and impart their high latent energies completely to the chamber walls.

The above consideration is of value for rockets operating only in space. For rockets destined to operate within the atmosphere of the Earth, the minimum applicable chamber pressure is limited at a still higher value, i.e., at several atmospheres, since the nozzle exit pressure cannot fall very far below the pressure of the surrounding gases, and in consequence the drop between the pressures in the chamber and in the nozzle exit area can become too small to furnish the desired high exhaust velocities.

It thus has been shown that the energy concentrations which can be utilized in rockets have an upper limit. It must be added that the position of this limit, which may be estimated approximately, is probably higher than the highest values attainable through simple chemical combustion processes, i.e., values of an order of magnitude of 10^{+4} k.cal./kg., and is lower than the values realized by nuclear reactions which are several powers higher in their turn. In order to utilize the range between the energy concentrations of the typical chemical combustion rockets with moderate exhaust velocities and those of not yet realized true nuclear rockets, it is intended as we know, in the so-called nuclear-thermal rockets, to add to the energy carrier (i.e., the particles having a high energy content and a negligibly small mass, m), in the chamber of the rocket, a mass carrier (i.e., particles having a large total mass, M, and a negligibly small energy content), in order to obtain the desired and just still tolerable, chamber enthalpy, $H_o \simeq H \cdot M/(m + M)$. This mass carrier we call the working fluid.

Since the individual physico-chemical properties of such a working fluid, e.g., molecular weight, faculties of dissociation and ionization, specific heats, emissive power, etc., co-operate in a somewhat complicated manner with the variables of thermodynamic state, i.e., pressure and temperature, for a given degree of heating, it is worth while examining the conditions for maximum exhaust velocities of certain interesting materials, by means of individual and sufficiently extensive enthalpy-entropy diagrams of these materials.

Thus, the following considerations are neither concerned with the kind of energy-producing nuclear reaction, nor with the mechanism by which the nuclear energy is imparted to the working fluid. In this respect it is merely assumed that the working fluid can be heated uniformly to any desired temperature by an energy source of infinitesimal mass.

Object of the investigation is, rather, the thermodynamic behaviour of certain working fluids, under the limiting assumptions that the combustion gases with a uniform temperature in the chamber of the rocket are at all times and regions in complete thermal and chemical equilibrium and undergo expansion at constant entropy, i.e., that accommodation to the varying thermodynamic states is accomplished without any relaxation of every degree of freedom, including chemical composition. The examination of the really possible deviations from this assumption and the influence of non-uniform heating of the working gas, respectively, of a non-homogeneous temperature distribution in the chamber, are reserved for a later study. As an example of

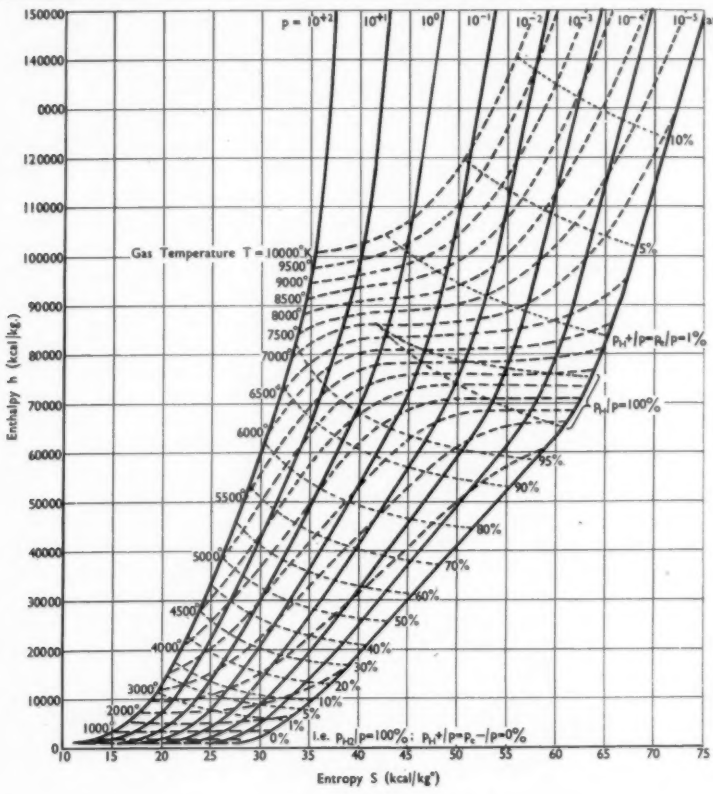

FIG. 1

a working fluid we have chosen ordinary hydrogen which is considered a favourite among the suitable working gases for nuclear-thermal rockets.

Fig. 1 shows the enthalpy-entropy diagram for hydrogen in the pressure range of between 10^{-5} and 10^{+2} atmospheres total pressure and in the temperature range from 0 to 10,000° K. Depicted are isobars, isotherms and the chemical composition according to the following equations

$$\left(\frac{p_H}{p}\right)^4 - 2\frac{K_{pH}}{p}\left(\frac{p_H}{p}\right)^3 + \frac{K_{pH}}{p}\left(\frac{K_{pH}}{p} - 2\right)\left(\frac{p_H}{p}\right)^2$$

$$-2\left(\frac{K_{pH}}{p}\right)^2\left(1 + 2\frac{K_{pH^+}}{p}\right)\frac{p_H}{p} + \left(\frac{K_{pH}}{p}\right)^2 = 0$$

$$\text{and } \frac{p_{H^+}}{p} = \sqrt{\frac{p_H}{p} \cdot \frac{K_{pH^+}}{p}},$$

where p means the total (overall) pressure, p_H and p_{H^+} the partial pressures of H and H^+ respectively, and K_{pH} and K_{pH^+} respectively the equilibrium constants for the dissociation process ($H_2 \rightarrow 2H$) and for the ionization process ($H \rightarrow H^+ + e^-$).

It can be seen that the temperature ranges for purely dissociating and for purely ionizing states approach each other with increasing pressure, until they finally merge at more than 1 atmosphere.

The basic calculations for this diagram were carried out according to the equations for ideal gases, i.e., without Van der Waal's corrections, as follows—

$$S_{(p,T)} = \frac{p_H\left[S_0^H + \int_0^T C_p^H \frac{dT}{T} - AR\ln\frac{p_H}{p}\right] + p_{H_2}\left[S_0^{H_2} + \int_0^T C_p^{H_2} \frac{dT}{T} - AR\ln\frac{p_{H_2}}{p}\right]}{\left(p - \frac{1}{2}p_H - \frac{3}{2}p_{H^+}\right)M_{H_2}}$$

$$+ \frac{p_{H^+}\left[S_0^{H^+} + \int_0^T C_p^{H^+} \frac{dT}{T} - AR\ln\frac{p_{H^+}}{p}\right] + p_{e^-}\left[S_0^{e^-} + \int_0^T C_p^{e^-} \frac{dT}{T} - AR\ln\frac{p_{e^-}}{p}\right] - pAR\ln p}{\left(p - \frac{1}{2}p_H - \frac{3}{2}p_{H^+}\right)M_{H_2}}$$

$$H_{(p,T)} = \frac{p_H\left[H_0^H + \int_0^T C_p^H\,dT\right] + p_{H_2}\left[H_0^{H_2} + \int_0^T C_p^{H_2}\,dT\right] + p_{H^+}\left[H_0^{H^+} + \int_0^T C_p^{H^+}\,dT\right]}{\left(p - \frac{1}{2}p_H - \frac{3}{2}p_{H^+}\right)M_{H_2}}$$

$$+ \frac{p^{e^-}\left[H_0^{e^-} + \int_0^T C_p^{e^-}\,dT\right] + \frac{1}{2}(p_H + p_{H^+})Q_0^{\text{Dis}} + p_{H^+}Q_{0\ \text{Ion}}}{\left(p - \frac{1}{2}p_H - \frac{3}{2}p_{H^+}\right)M_{H_2}}$$

The partial pressures of the hydrogen atoms p_H, the atomic ions p_{H^+}, the free electrons p_{e^-} and the neutral hydrogen molecules p_{H_2} were taken into account. The partial pressures of the ionized hydrogen molecules $p_{H_2^+}$ proved to be negligibly small under conditions of equilibrium, since dissociation is practically completed *before* any significant ionization of H_2 sets in. The ionization was calculated in a way permissible with low plasma field strengths, inasmuch as the specific heat of ions and electrons, were derived in the same manner as those of monatomic non-ionized gases, and as the Law of Mass Action was applied in the form valid for the dissociation of ideal gases, viz. without the introduction of activities instead of concentrations. Actually, the maximum partial pressure of ionized hydrogen shown in Fig. 1 is something

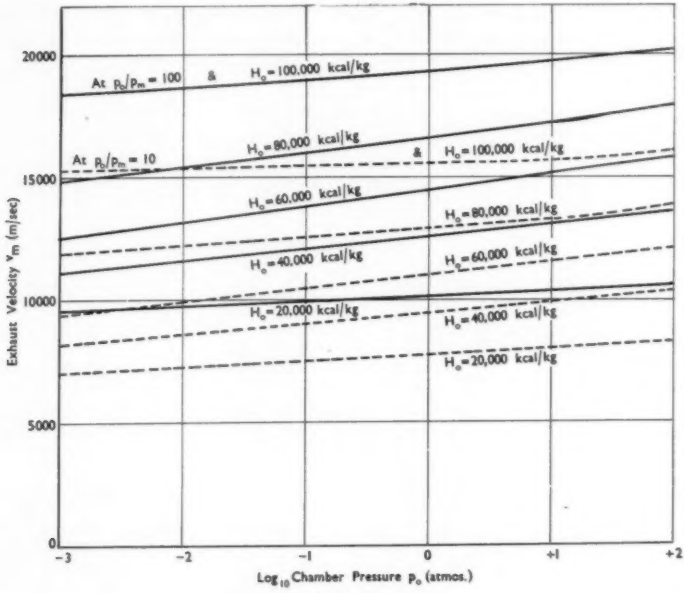

FIG. 2

over 10 per cent. In the calculation of Planck's partition function Z and the quantities derived therefrom, such as specific molecular heats C_p, equilibrium constants K_p, enthalpies H and entropies S, the degrees of freedom of translation, of rotation and of vibration (including the influence of anharmonicity and the stretching effect on the vibrational energy of the H_2 molecules) were taken into account, as well as the shares of the dissociation and ionization energies $Q_{0\,Dis}$ and $Q_{0\,Ion}$. The proportion of electron excitation energy in the partition function proved to be negligibly small due to a minimum value for the relevant characteristic temperatures of about 118,000° K.

If we assume isentropic expansion, the realizable flow velocities in the exit cross section of the nozzle—as is well known—can be deduced from the entropy-enthalpy diagram for any given expansion conditions *po/pm* between the chamber and the exit pressures and for all values of chamber pressure and chamber enthalpy, which are shown in Fig. 1. Since this exit flow velocity v_m is equal to the effective exhaust velocity, c, which occurs directly in the basic rocket equation, apart from a small additional factor determined mainly by the nozzle configuration, it may be taken—in thermodynamic considerations—as a measure for the realizable flight performances.

Fig. 2 shows the exit flow velocities in function of chamber pressure for a few given enthalpies. This is the kind of graphic representation, as it was used and justified until now for rockets which derive their energy from chemical combustion processes with calorific values determined by the materials and

independent of thermodynamic chamber conditions. This representation further resulted from the use of constant expansion ratios, i.e., for rockets with more or less geometrically similar nozzles. For comparison sake, the sets of curves for the two expansion ratios $p_o/p_m = 100$ and 10 are plotted. Each system contains the iso-enthalpies for the integer powers ranging between 10^4 and 10^5 k. cal./kg.

The diagram shows that for geometrically similar rockets and heating up to constant chamber-enthalpy, the obtainable exit flow velocities increase very slightly with the chamber pressure. This may be explained by the fact that at the highest chamber pressure the highest percentage of the constant enthalpy is given initially in the form of pure sensible heat and thus, at constant expansion ratio, yields the greatest conversion into kinetic energy, or—speaking mathematically—the greatest quasi-x values for the expansion. If the total enthalpy of gas mixture ceases to have an upper limit determined by its individual calorific value, and on the contrary it becomes possible to heat any gas mixture to any desired enthalpy in the chamber, a more suitable parameter should be used in place of the iso-enthalpies. This parameter characterizes the new, henceforth binding, technical upper limit to the usable chamber enthalpies, namely the heat transfer to the walls bounding the heated gas, or a function of the thermodynamic state values of the gas, which is unequivocally related to the heat transfer. Of the two components of heat transfer, radiation and convection, one or the other will predominate according to the prevailing conditions (temperature, pressure, radiation properties of the working gas). Since the two types of heat transfer are subject to entirely different laws, they will be treated separately.

Where radiation determines the magnitude of heat transfer, a power of the temperature (the fourth, for continuous radiation under equilibrium conditions!) will determine the approximate value of the heat transfer effectively enough—assuming an optically infinite thickness of the gas layer. For non-quantitative, comparative optimum analysis within one and the same given working gas, the temperature itself will be a sufficient parameter. Strictly speaking, the functional coherence of the gas radiation with T^n only really applies to radiation, in which the number and width of emission lines do not vary in the comparative region of pressure and temperature. This, however, will not come true in a *reacting* gas, the various components of which have their own individual spectra. Neither does it apply even to *uniform* gases *in those* temperature and pressure *ranges*, where a broadening of the lines may be expected through collision quenching, through thermal Doppler effect, or through atomic electro-magnetic fields. Moreover, the radiation must be particularly strong locally, where continuous spectra appear, i.e., in ranges where the dissociation or ionization gradient is at its maximum; for instance when $p_H/p = 0{\cdot}73$, viz. $p_{H^+}/p = 0{\cdot}37$. This increase must become relatively more apparent at low pressures, just like that of the local specific heats.

For the purpose of gaining a general impression we have contented ourselves, in the meantime, with the simplest approximation.

This yielded *Fig.* 3 which—for co-ordinates and expansion ratios similar to those in Fig. 2—shows chamber-temperatures, i.e., the entire thousands between

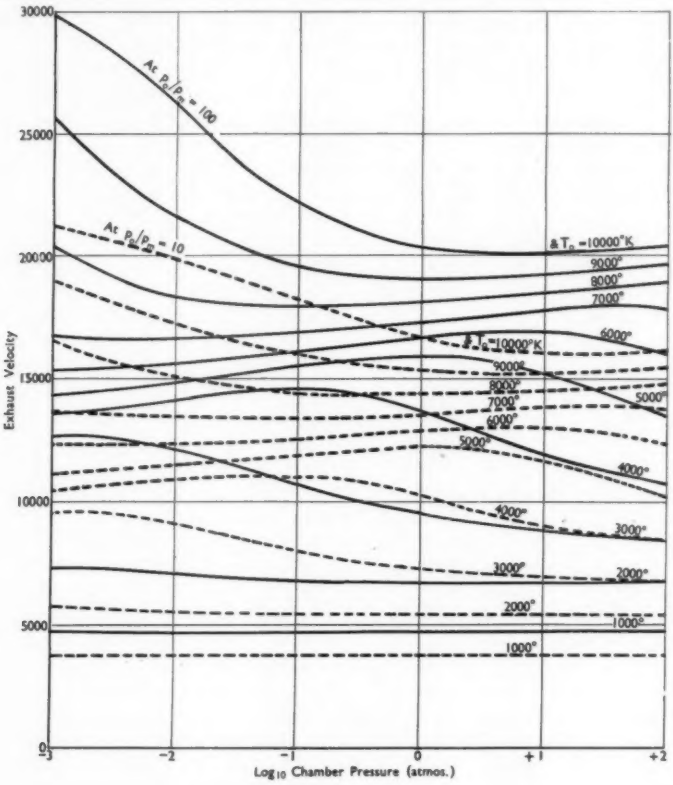

FIG. 3

$T_0 = 10^3$ and $10^{4°}$ K. Next, it will be noticed there that the exit flow velocities indicate definite maxima at certain chamber pressures and for all isotherms above a certain level, below which no pressure dependent energy transformations take place, and from that the resulting exit flow velocities do not vary with pressure. On account of the power law of radiation, these maxima will become still more pronounced in reality, while retaining their abscissa value p_o. Furthermore, their abscissa value p_o is not sensibly dependent on the expansion ratio, but only on the value of the flame temperature, i.e., depending on the conditions of thermodynamic state within the chamber. A comparative glance at the phase diagram in Fig. 1 shows that, with respect to the velocity maxima in their dependence upon chamber temperature and pressure, a first local maximum value meets the knee of the isotherms at about 98 per cent. dissociation; a second, which—in the range of states under consideration in the diagram—is but faintly recognizable, the corresponding knee when ionization is almost

complete. Accordingly, the velocity maxima move with rising flame temperatures, with accompanying flattening, to higher chamber pressures. Only beyond 8,000° K., the velocity maximum due to dissociation would attain the region of $p_o = 100$ atmospheres. At such a chamber temperature and below $p_o = 10^{-2}$ at, however, the absolute values of spurs of the second maxima caused by ionization already predominate. Rockets with chamber pressures greater than 100 atmospheres therefore seem uninteresting in this respect. The position of the maximum exit flow velocities in the system of co-ordinates of the state quantities in the chamber signifies physically that, at almost complete dissociation or ionization, enthalpy at a given temperature, i.e., the mean specific heat of the gas, is at its highest and, at the same time reducible by even the least drop in pressure, that means convertable into kinetic energy.

So much for the optimum conditions in rockets with a preponderance of radiation heat. Since one would prefer as working fluids highly diathermal gases which will only begin to radiate at extremely high pressures and temperatures, the examination of the conditions in preponderantly convectional heat transfer will be more decisive.

In the field of free molecular flow of the combustion gases (which is unfruitful so far as rocketry is concerned), i.e., when in comparison with chamber dimensions the molecular mean free paths are very great, since

$$Q = (\gamma \bar{w}/4) \cdot [A w_x^2/2g + (C_p - AR/2)\, T/M]$$

(where γ = spec. wt. of gas; $\bar{w}$ = mean molecular velocity of gas, vide refs. 2, 12, 13, 14, 15, 16) the energy transport per second, Q, to the unit area of the walls, and for flow parallel to the walls with very low velocity w_x, is

$$Q = \text{constant} \cdot p \cdot H \cdot (M/T)^{1/2},$$

in terms of pressure, enthalpy, molecular weight and temperature in the combustion gas.

At comparably negligible wall temperatures, this energy will be wholly absorbed by the wall.

Strictly speaking, the question of whether, at each contact of a gas particle with the wall, the total enthalpy is transferred thereto, is still debatable in the field of gas kinetics. First it must be remembered that, of course, every particle arriving will surely impart to the wall its total translational energy thus that, however, by an integral consideration of a sufficiently extended gas space, 4/3 of the mean energy of translation of the gas is transferred per second to unit area of the wall, because just in the translational motion—as the only degree of freedom—the magnitude or the energy transmitted is coupled with the number of collisions per second, i.e., that the fast particles with a high translational energy collide more frequently than their distribution in space warrants. The increase of ART/2, as it happens, represents exactly one half of the compression work contained in the enthalpy, so that in unrestricted transfer of the internal energies to the wall, the heat transfer per second would be proportional to $(H - \text{ART}/2)$. Apparently, however, the stay of the gas particles in the wall is not always sufficient for complete quenching of vibrational energy. Furthermore, recombination of the dissociated particles at the wall will be rare

due to a lack of recombination partners; on the other hand, recombination of ions with the wall electrons is possible. All internal energy not imparted to the wall will, however, hardly be converted into kinetic energy of flow as a result of lacking collisions with other particles during passage through the nozzle after leaving the chamber, so that one could calculate the whole process formally by means of a quasi-enthalpy, which is smaller than the actual enthalpy of the combustion gas.

In the field of *thermodynamic* flow with moderately large mean free paths, one obtains with the aid of the Nusselt Number for turbulent flow along flat plates:[12]

$$Nu = 0{\cdot}075\ (Pe)^{0{\cdot}75} = 0{\cdot}075\ (w_x{\cdot}l{\cdot}c_p{\cdot}\gamma/\lambda)^{0{\cdot}75}$$

and again with negligible wall temperatures, in particular:

$$Q = 0{\cdot}075\ (w_p{\cdot}\gamma{\cdot}c_p)^{0{\cdot}75}\ (\lambda/l)^{0{\cdot}25}{\cdot}T$$

$$= 0{\cdot}075\ (w_x{\cdot}\gamma{\cdot}H)^{0{\cdot}75}(\lambda T/l)^{0{\cdot}25}$$

[k. cal./m^2s] for heat transfer from a homogeneous combustion gas to the chamber walls. w_x denotes again the velocity of flow and l the axial length of the chamber. If one substitutes in this equation for the thermal conductivity, λ, with appropriate sufficiently accurate approximation based on the Chapman-Enskog theory for rigid spherical molecules, the expression

$$\lambda = (1{\cdot}522\ c_{v_{trans}} + c_v)\ \eta$$

$$= (1{\cdot}522\ c_{v_{trans}} + c_v)\ \frac{0{\cdot}499}{2\sqrt{\pi}}\ (m/\phi)\ \sqrt{kT/m}$$

[k. cal./$m°s$] (where η is the viscosity, k Boltzmann's constant, c_v the specific heat at constant volume per unit mass, $c_{v_{trans}}$ the translational component of the latter, and m/ϕ the molecular mass, m, divided by the gas kinetic effective cross section, ϕ) we obtain

$$Q = z\left[\frac{w_x\bar{w}\gamma H}{l^{1/3}(m/\phi)^{-1/3}}\right]^{3/4}.$$

This expression—apart from the dimensionless magnitude,

$$z = 0{\cdot}075\left[\frac{0{\cdot}499\ \pi\,(1{\cdot}522 c_{v\ trans} + c_v)\ m}{32\sqrt{2k}}\right]^{\frac{1}{4}}$$

as well as the experimental constants w_x and l—is just proportional to a power of the product of the specific weight of the gas, γ, the gas enthalpy, H, the mean molecular velocity, $\bar{w}$, and the cube root of the specific molecular mass, m/ϕ. It is therefore possible—similarly as for free molecule flow conditions—to write again:

$$Q = \text{const.}\ [pH(M/T)^{\frac{1}{2}}(m/\phi)^{\frac{1}{3}}]^{\frac{3}{4}},$$

where the constant here in its characteristics is different from that for free molecule heat transfer, namely,

$$\text{const.} = 0{\cdot}0324\left[\frac{1{\cdot}522\ C_{v\ trans} + C_v}{R}\cdot\frac{w_x^3}{l}\right]^{1/4}.$$

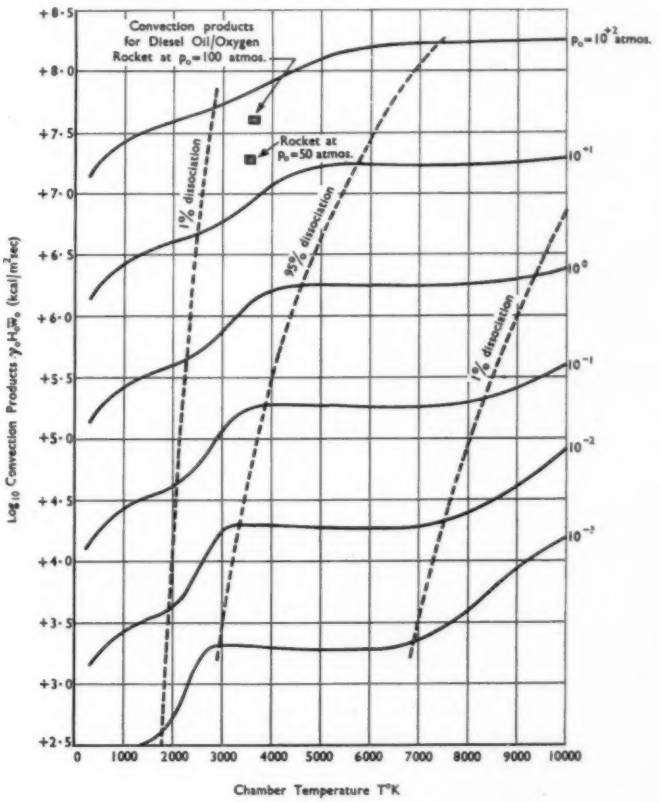

FIG. 4

Ignoring the specific molecular mass, m/ϕ—which is not easily expressed exactly in figures—occurs at a comparatively low power, and the omission of which means that the calculated convection product, e.g., for very light gases and for ions with very high effect cross section becomes, comparatively, somewhat too great—there remains in the gas kinetic as well as in the gas dynamic regimes of flow a dependence of the convective heat transfer on the same product of thermodynamic magnitudes $pH \cdot \sqrt{M/T}$, or—more demonstrative—$\gamma H \bar{w}$. One may thus consider the convective heat transfer in general as a function of the product of the molecular enthalpy $\gamma H/N$ of the gas particles, their number N perunit volume and their mean thermal velocity $\bar{w}$, where the last two factors denote the frequency with which the energy of the individual particles is imparted to unit area of the wall per second. The function itself alters its character according to the type of flow regime considered. For comparison

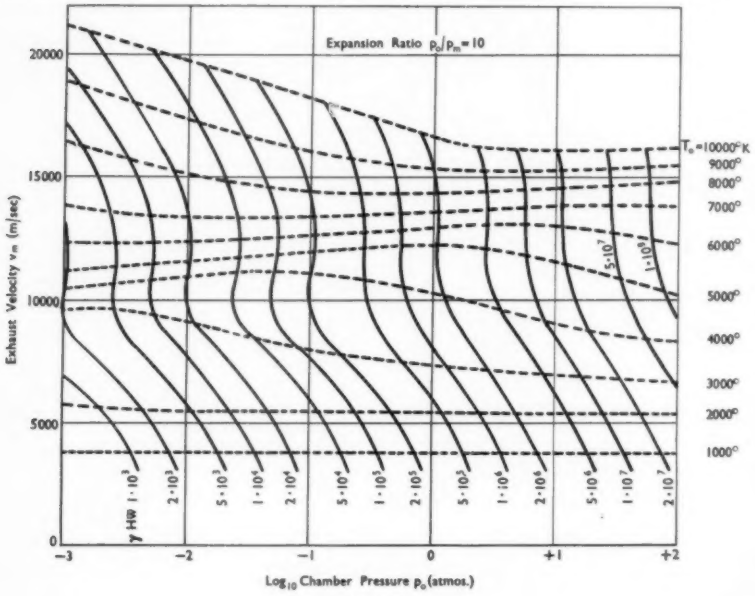

FIG. 5

of the influences of the thermodynamic quantities, p, T and M within the same flow regime, it will therefore suffice to consider, instead of the convective heat transfer itself (analogous to the introduction of T in place of radiation heat transmission), the parameter $\gamma H\bar{w}$, which will be called the convection product for short. For gas mixtures follows the summation of the product over the partial values, $\gamma H\bar{w} = \Sigma_i(\gamma_i H_i \bar{w}_i)$.

This convection product is shown quantitatively in *Fig.* 4 for hydrogen in its dependence on pressure and temperature. One can see in each of the curves for constant gas pressure that the convective heat transfer increases considerably in the region of excitation of molecular vibration and of dissociation. The convection product increases to approximately ten times its value at normal temperature in these ranges, and from then on until the beginning of ionization it remains practically constant. At constant mean molecular weights and specific heats, the convection product rises along an isobar only with the square root of the temperature. In regions of state where a rise in temperature is not accompanied by a variation of molecular weight or specific heat—as for instance in non-reacting monatomic gases—one may increase the chamber temperature and hence the exhaust velocity—keeping the chamber pressure constant—without any considerable change in convective heat transfer to the walls of the chamber. It is only with the beginning of ionization that the convection product increases strongly, as a result of the high contribution of free electrons to the heat transfer.

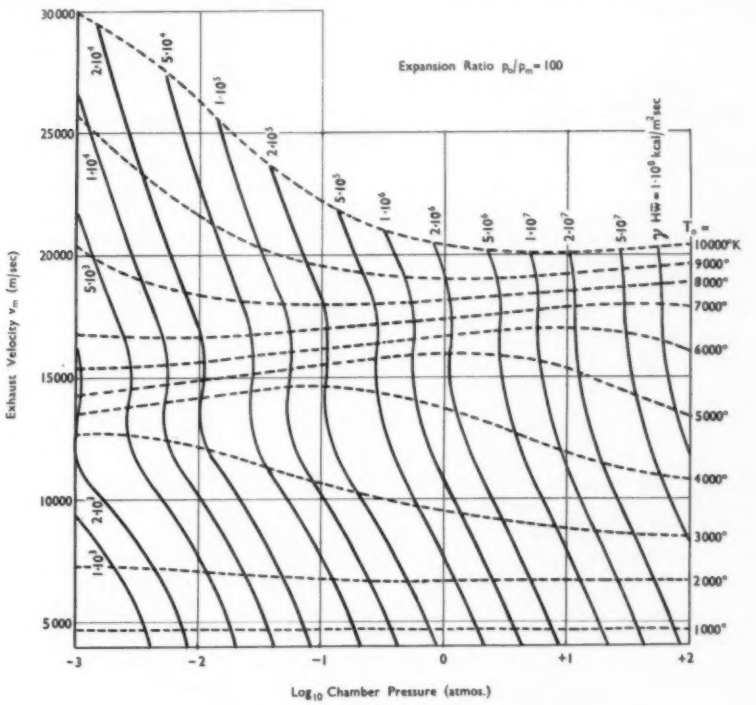

FIG. 6

In Fig. 4, the small shaded square shows the approximate value of the convection product for a kerosene-oxygen rocket at 50 atm. chamber pressure and stoichiometric combustion. Heat transfer encountered in such an operation could already definitely be controlled at the beginning of the war at the proving ground at Trauen (*l.c.*[2], p. 34). There, also, the then current peak values for heat transfer were achieved for short operation times at 100 atm. and with the same rockets. The corresponding convection product is marked by the tip of an arrow in Fig. 4. If one considers that of the heat transfer appearing experimentally in the combustion of hydrocarbons, an appreciable portion is radiative transfer, the shares representing the convective transfer shown in Fig. 4 may be taken as surely controllable. They would appear in the chamber of the low emission hydrogen rockets, for instance, at 10,000° K. and about 10 atm., and probably constitute the preponderant part of the heat transfer.

In *Figs.* 5 and 6, where the dotted isotherms repeat the results of Fig. 3 disconnected with an expansion ratio of $p_o/p_m = 10$ and of 100, the solid-line groups represent curves of equal convection products $\gamma H\bar{w}$, in dependence on chamber pressure and exit flow velocity of the rocket. For $\gamma H\bar{w}$, values

between 10^{+3} and 10^{+8} k. cal./m.²s have been entered, where the last value also represents the upper limit of the heat transfer actually controllable. Both figures confirm that for constant convection product, i.e., constant convective heat transfer, the exhaust velocity increases rapidly with decreasing chamber pressure. With a convection product of $2 \cdot 10^7$ k. cal./m.²s—representing approximately the already technically controlled heat transfer of a 50 atm. kerosene-oxygen rocket—the exhaust velocity of a nuclear working fluid rocket, having hydrogen as a working fluid and an expansion ratio of 10—assuming equilibrium—would only be 5,500 m./sec. at 50 atm. chamber pressure and 2,000° K. At equal convection and expansion conditions, the exhaust velocity would however rise to 16,100 m./sec., if the pressure in the chamber is reduced to 10 atm. with a simultaneous temperature increase to 10,000° K. The relevant comparative figures at equal convective heat transfer but at an expansion ratio of 100, are

$$p_o = 50 \text{ atm.}; \; T_o = \; 2{,}000° \text{ K.}; \; w_m = \; 6{,}700 \text{ m./sec.}$$

$$p_o = 10 \text{ atm.}; \; T_o = 10{,}000° \text{ K.}; \; w_m = 20{,}100 \text{ m./sec.}$$

At equal convective heat transfer, a reduction of pressure to 1/5 of its initial value therefore yields in both cases an increase in flow velocity in the nozzle exit of about three times its initial value. It is possible, with still lower heat transfer and adequate considerably smaller chamber pressures, to achieve exit flow velocities up to 30,000 m./sec. with hydrogen, where the mean free paths of the gas in the chamber still remains in the order of 1 mm., for hydrogen atoms and 5 cm. for protons and electrons.

Should one desire to apply the results obtained in the investigation of hydrogen as working fluid, to other substances, in order to evaluate their behaviour as working gases, the following directions are indicative. The variability of the exit flow velocities with the absolute chamber pressure at equal pressure drop ratios and equal *chamber temperatures*, is only given in such working gases whose composition varies with pressure, i.e., which dissociate or ionize with change of mole-number. If the comparison is made with iso-*enthalpies* in the chamber instead of isotherms, the exit flow velocities will become even pressure-dependent already with pure *temperature* dependence of the specific heats of the mixture, e.g., in the range of varying vibrational specific heat. Wholly independent of the magnitude of the chamber pressure are on the other hand, at constant pressure ratios, the exhaust velocities of working gases, which have constant mean molecular weights and constant specific heats of the mixture in the observed range of state—especially the exhaust velocities of monatomic, non-radiating gases. The magnitudes of the attainable exit flow velocities are here *constant* for all chamber pressures, and determine for various working gases at a given chamber temperature—as is well known—by the molecular weight as well as the specific heat:

$$w^2_m = 2g/A \cdot H_0[1 - (p_m/p_0)^{AR/C_p}]$$

$$= 2g/A \cdot C_p \cdot T_0/M \cdot [1 - (p_0/p_m)^{AR/C_p}].$$

In this way, helium ($M = 4{\cdot}003$, $Cp = 5/2\ AR$) which at 10,000° K. and 10^{-3} atm. has an ion concentration of only 0·9 per cent. or at 10^{-5} atm. already 8 per cent., and whose characteristic temperatures for electron excitation from the ground state exceed 246,000° K.—with chamber temperature of $T_o =$ 10,000° K. and any chamber pressure above 10^{-3} atm.

with an expansion ratio of $p_o/p_m = 100$ would yield an exhaust velocity of

$$w_m = 9{,}348 \text{ m./sec.},$$

and with $p_o/p_m = 10$, an exhaust velocity of $w_m = 7{,}906$ m./sec.

If hydrogen would remain entirely monatomic ($M = 1{\cdot}008$, $Cp = 5/2\ AR$) in all ranges of thermodynamic state its exhaust velocity would be

$w_m = 18{,}776$ m./sec. at $T_o = 10{,}000°$ K. and any given chamber pressure—

while in reality it rises from 20,415 m./sec. at $p_o = 100$ atm. to

$$29{,}898 \text{ m./sec. at } p_o = 10^{-3}\phi.$$

Even then, the first figure is still higher than the corresponding value for helium.

In contrast to the behaviour of the exit flow velocities with isotherms and iso-enthalpies, the exit flow velocities in the case of constant *convection* product in the chamber *always* increase rapidly with decreasing chamber pressure, quite insensitive of the particular thermodynamic properties of the working fluid. This does not mean that for all materials with given values for convection product, chamber pressure and expansion ratio, the exit flow velocities are all equally great. Under these conditions the monatomic inert working gases achieve the highest exhaust velocities, with relatively high heating temperatures. Therefore, with the above assumption, gas temperatures and exhaust velocities of hydrogen are lower for all practical chamber pressures, than their corresponding values for helium, namely, e.g., for $p_o/p_m = 100$ and $\gamma H\bar{w} =$ $4{\cdot}26 \times 10\ p$ k. cal./m.^{2}s are about 2,000° K. and 7,050 m./sec. for hydrogen as compared to 10,000° K. and 9,350 m./sec. for helium. The low convective heat transfer of the working gas hydrogen at 10^{-3} atm. and 10,000° K., would be reached by helium only at a temperature so high that helium would already have become ionized, and radiate to such an extent that the radiation component of heat transfer could no longer be neglected. The suitable absolute values of hydrogen of 30,000 m./sec. exhaust velocity for a convection product of $1{\cdot}5 \times 10^4$ k. cal./m.^{2}s, will never be reached, for example by helium—despite the superiority of helium with respect to heat transfer rates in the range of moderate exhaust velocities.

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* Die Zahlenwerte für die verschiedenen physikochemischen Konstanten, wie charakteristische Temperaturen θ, statistische Gewichte g, Reaktionswärmen Q_0 usw. wurden dabei der neueren Literatur entnomen[8,9,10].

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SPACE TRAVEL AND FUTURE RESEARCH INTO THE STRUCTURE OF THE UNIVERSE

By Dr. Ir. J. M. J. Kooy

SUMMARY

According to the leading cosmological theories, intergalactic space is spherical, the radius of curvature varying with time. According to Eddington,[1] there are only two ways to account for the large speeds of recession of the spiral nebulæ, they are produced either by a scattering force (called "cosmical repulsion," predicted by the general theory of relativity) or the receding velocities now observed have existed from the beginning. For completeness, he advances as a third possibility an oscillating model of the universe.[2] To-day the question which of these ways must be taken is still unsettled. As Eddington remarked, more than a quarter of a century ago,[3] our best hope for further progress is some quantitative test. In this paper an extensive programme of astronomical observation is indicated in order to test the alternative cosmological theories.

In order fully to carry out such a programme, an extra-terrestrial observatory will be indispensable.

1. Introduction

The ultimate result of space-flight is and will always be to obtain more knowledge of the universe outside the Earth. We hope to reach the Moon and even the nearer planets, but these trips to not mean that space will be conquered. They will be quite insignificant in comparison to the distances with which we have to deal in intergalactic space. But nevertheless, space travel will be of the utmost importance for research into this immense universe. The restrictions due to irregular diffraction, selective absorption and

luminosity of the upper layers of the atmosphere will be eliminated. Further, in the case of telescopes freely coasting in space, there will be no more deformation due to gravitational forces. Much larger apertures can be applied and the resolving power can be increased almost indefinitely, and also the whole spectral range will be at our disposal.

2. Short Survey of Principal Cosmological Ideas

As is well known, the large system of spiral nebulae, constituting for us the visible universe, expands in such a way that any object of this system can be considered as the centre of expansion, whereas the speed of recession increases proportionately with distance. According to Hubble, taking into account the newest estimates, the speed of recession increases at a rate of about 55 km./sec. per million light-years. In order to explain this cosmological phenomenon of expansion, different theories have been advanced, of which those of Eddington and Lemaître are the most prominant. Both Lemaître and Eddington assume that intergalactic space is spherical, while the radius of curvature increases with time. According to Eddington's view, some 90 thousand million years ago the radius of space curvature, and thereby space itself, was much smaller than it is now, and the primeval matter was practically evenly distributed. Random motions in this primeval matter caused local concentrations. Thereby the original state of rest was disturbed; the system began to split up in smaller parts, receding from one another, by which spherical space itself increased. The smaller patches then developed into the stellar systems we now see. These views are in accordance with the general theory of relativity according to which a repulsive scattering force must be added to the gravitational attraction of bodies. As Eddington points out[4] this cosmical repulsion is proportional to the distance and is overwhelming at intergalactic distances. Lemaître assumes on the other hand that the primordial matter was concentrated in one gigantic atom, filling the whole of original space. The expansion of the universe began by explosion of this enormous atom and the different parts in which it was split up obtained speeds—which they have retained up to the present. By their moving apart spherical space itself increased, the radius of space curvature growing proportional with time. Accepting the value of Hubble's constant mentioned above, the initial explosion then occurred some 5,500 million years ago. Now according to the latest results of observation the speeds of recession of the most remote nebulae appear to be smaller than expected on account of the conventional supposition of proportionality of these speeds with the distance. This feature seems to reconcile with the idea of an expansion against inward directed forces, corresponding with an oscillating model of the universe.

3. General Consideration of the Possibility of Determining by Observation the Radius of Space Curvature and its mode of Variation with Time

We now can pose the question: Will it be possible for mankind with its (in cosmological terms) undoubtedly extremely short span of life to find out empirically, that is by astronomical evidence, how the radius of space curvature varies in periods, enormously surpassing the momentary existence of the human

race on Earth? As a first attempt to attack this problem, we can begin by writing most generally for the radius of space curvature:

$$R = F(t) \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (1),$$

in which t is the time co-ordinate. [At first view it seems a contradition to the theory of relativity to introduce a time co-ordinate, which avails for the whole universe. But, as Eddington observes:[5] "just as a frame of space and time defined with reference to the Sun is appropriate for dealing with problems relating to the solar system, so a frame defined with reference to the matter of the universe as a whole is also appropriate for dealing with the universe as a whole. In a uniform spherical world (a mathematical simplification of the real world) the frame appropriate to the universe as a whole is also appropriate to every part of it; thus the usual multiplicity of frames of space and time is suppressed. The principle of relativity is that one man's frame is as good as another's; it is not upset by imagining an ideal world (the uniform spherical world) in which every man has hit on the same frame."]

According to (1) the rate of increase of R becomes:

$$U = \frac{dR}{dt} = F^1(t) \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (2).$$

If the geodesic arc, passed through by a light wave in the time interval dt, be dX, then:

$$cdt = RdX = F(t)dX,$$

if c be the speed of light. Using the light-year as unit of length and the year as unit of time, $c = 1$; we then obtain:

$$\int_{t_1}^{t_0} \frac{dt}{F(t)} = X \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (3),$$

in which $t = t_0$ denotes the present and $t = t_1$ the instant of time at which the light started of an image we now receive from a celestial object at a geodesic *arc* distance X. *Now let us suppose that the speeds of recession of the spiral nebulae are only due to the expansion of spherical space.* (*Superimposed motions are disregarded.*) *Hence an object at geodesic arc distance X continues to be at this arc distance X, while the radius of space curvature R increases.* The speed of recession of a stellar system or of a cluster of stellar systems at geodesic arc distance X then becomes:

$$V = F^1(t_1)X \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (4)$$

in which X is a function of t_1, defined by (3). Writing (3) in the form:

$$X = \psi(t_1) \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (3a)$$

and substituting (3*a*) in (4), we obtain:

$$V = F^1(t_1)\psi(t_1) \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (5).$$

If we now assume, by means of trial, a certain graphical course for $R = F(t)$,

we can also determine the course of $X = \psi(t)$ and consequently solve t_1 from (5), V being given.

Further, if δ be the angle at which a spiral nebula of real diameter a is seen in spherical space with radius R, the arc distance X of the nebula follows from:

$$\cos\gamma = \cos^2 X + \sin^2 X \cos\delta, \text{ in which } \gamma = \frac{a}{R}$$

Hence:

$$\sin X = \sqrt{\frac{1-\cos\gamma}{1-\cos\delta}} = \frac{\sin\frac{\gamma}{2}}{\sin\frac{\delta}{2}} \quad .. \quad .. \quad .. \quad .. \quad (6)$$

Now in spherical space the propagation of a ray of light occurs along a geodesic arc. Hence, if we consider two different rays of light in spherical space, the propagation occurs along two different arcs. If the spherical space expands during the time of propagation, both rays continue to travel along the same geodesic arcs, while their radius equal to the radius of spherical space increases. Thus the angle included by the two geodesic arcs is not altered by the expansion of space, so that also the angle at which a certain object is seen in spherical space is the same as the angle at which this object would have been seen if no expansion had taken place during the period of light travel. Hence, if in expanding spherical space, δ be the angle at which a spiral nebula of real diameter a is seen, the geodesic arc distance follows from (6), in which $\gamma = a/R_{t_1}$, R_{t_1} being the radius of space curvature at time t_1, at which the light we now receive started from the nebula. Substituting (1) and (3a) in (6), we then obtain:

$$\sin\psi(t_1) = \sqrt{\frac{1-\cos\frac{a}{F(t_1)}}{1-\cos\delta}} = \frac{\sin\frac{a}{2F(t_1)}}{\sin\frac{\delta}{2}} \quad .. \quad .. \quad .. \quad (7).$$

Now the course of the function $R = F(t)$ must be adjusted in such a way, that (7) is satisfied for a number of spiral nebulae as large as possible, of which a as average value must be estimated, whereas δ and V must be measured (V can be measured spectroscopically). The value t_1 which must be substituted in (7), then follows from (5). On the other hand, if E be the absolute luminosity and E^1 the (corrected) apparent luminosity of a stellar system at geodesic arc distance X, we have:

$$E^1 = \frac{E}{4\pi R^2_{t_0} \sin^2 X} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (8)$$

in which $R_{t_0} = F(t_0)$ denotes the present radius of curvature.

[The quantity E^1 is the apparent luminosity *not reduced by red displacement*, whereas the apparent luminosity which is directly observed is smaller. By the red shift the energy of each light quantum will be reduced by decrease of frequency, whereas also the number of quanta received per second will be diminished on account of the radial speed of recession of the source of light.

Then if ν be the frequency of the emitted light and ν^1 the observed frequency, we obtain by the decrease of energy of each photon a weakening $\frac{\nu^1}{\nu}$, and by the decrease of the received number of quanta again a weakening $\frac{\nu^1}{\nu}$ of the apparent luminosity. Thus, if E^{11} be the observed apparent luminosity, we obtain:

$$E^1 = E^{11}\left(\frac{\nu}{\nu^1}\right)^2, \text{ whereas } \frac{\nu^1}{\nu} = \frac{1-V}{\sqrt{1-V^2}},$$

in which V is expressed in light-years/years.

$$\text{Hence:} \qquad E^1 = E^{11}\frac{1+V}{1-V}. \quad \Big]$$

Hence, if of the observed stellar systems quoted above also, average value of E can also be estimated and E^1 derived from measurement, the course of the function $R = F(t)$ must be adjusted in such a way that not only (7), but also (8) will be satisfied.

In practice we can gather a group of nebulæ, all having the same value for V and determine the average values of E^1 and δ over the specimens of the group. For E and a we then assume the same average value as for the stellar systems in the immediate neighbourhood of our own galactic system, the distances of which are approximately known. This procedure must then be carried out for different groups of spiral nebulae, corresponding with different values for V.

4. The Assumption of Linear Expansion

After having explained the general method of observational research, let us firstly apply this procedure to the model of Lemaître. In this case we have linear expansion, so that (1) becomes:

$$R = Ut \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (9),$$

in which U is a constant, being the constant rate of expansion of the radius of space curvature. The speed of recession of a celestial object at geodesic arc distance X then becomes:

$$V = UX \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (10).$$

Further we obtain by (3):

$$\int_{t_1}^{t_0}\frac{dt}{t} = UX, \text{ whence follows:}$$

$$t_1 = t_0 e^{-V} \quad .. \quad .. \quad .. \quad .. \quad (11)$$

Then further (6) and (7) become, in connection with (9), (10) and (11):

$$\sin\frac{V}{U} = \frac{\sin\frac{a}{2Ut_0e^{-V}}}{\sin\frac{\delta}{2}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (12)$$

$$\sin \frac{V}{U} = \frac{1}{2Ut_0}\sqrt{\frac{E}{\pi E^1}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (13)$$

whence follows:

$$\frac{1}{2Ut_0}\sqrt{\frac{E}{\pi E^1}} = \frac{\sin \frac{a}{2Ut_0 e^{-V}}}{\sin \frac{\delta}{2}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (14)$$

In (13) and (14) t_0 and U are the unknowns. Ut_0 can be solved numerically from (14) and consequently U can be found from (13). If then for different groups of spiral nebulæ (corresponding with different values for V) the same values for U and t_0 are found, this will be an indication that a linear expansion (expressed by (9)) is rather in accordance with the truth. If on the other hand (14) appears to have no solution, this will be an indication that the expansion is of non-linear type.

Now it is possible, assuming that linear expansion really occurs—at least in first approximation—to make an estimate of the results which can then be expected. As we have seen above, according to the latest estimate of Hubble's constant we obtain:

$$t_0 \approx 5,5 \times 10^9 \text{ years} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (15).$$

In order further to estimate U, we must make some reasonable supposition. If we accept the cosmological conception of Lemaître, we must suppose that the initial explosion must have been the most violent explosion ever possible, so that we may assume that at least some of the emitted patches practically approached the speed of light, retaining this speed up to the present. In that case the maximum speed of recession would be the speed of light.

On the other hand it is an open question if the observed speeds of recession must be seen as simply identical with the initial speeds of the different patches at the time of explosion. The whole idea of an absolute beginning seems irrational, only bound up with the restricted outlook of absolute space and time. Therefore let us assume most generally that the maximum speed of recession is equal to n, the speed of light being unity.

Then by (10), $U\pi = n$, or $U = \frac{n}{\pi}$ light-years/year .. .. (16).

Then the present radius of curvature becomes:

$$R_{t_0} = \frac{n}{\pi} \cdot 5{\cdot}5 \times 10^9 = n{\cdot}1752 \times 10^6 \text{ light-years.}$$

If the speed of recession approaches unity (the speed of light) the frequence of the received light falls to zero so that from an object corresponding with $V>1$ no light can be received. Hence the age of the oldest light we now observe is according to (11):

$$t_0\left(1-\frac{1}{e}\right)=5{\cdot}5\times10^9\left(1-\frac{1}{2{,}7}\right)=3465\times10^6 \text{ years.}$$

This oldest light then corresponds with the largest optical distance. This distance of the corresponding objects at the time of light emission then amounted to

$$\frac{\pi}{n}\frac{n}{\pi}\frac{t_0}{e}=\frac{5{\cdot}5\times10^9}{e}=2037\times10^6 \text{ light-years}$$

and has now (at the time t_0 of light reception) increased to 5500×10^6 years. Further it is of interest to remark that the total number of galaxies can be estimated when n is known. If the average distance of a galaxy to the nearest galaxy is about two million light-years, we have in average one galaxy in a volume of 8×10^{18} light-years. The volume of the present spherical space becomes $2\pi^2R^3_{t_0}$. Hence if $n=1$ we obtain as total number of galaxies:

$$\frac{2\times3{\cdot}14^2\times1752^2\times10^{18}}{8\times10^{18}}\approx13\times10^9.$$

The optical horizon, corresponding with $V=1$, or $X=\frac{\pi}{n}$, shrinks to a point if $n=1$, and thus only exists if $n>1$. Objects beyond this boundary are invisible. Nevertheless it is quite possible that influences of objects beyond the optical horizon on objects at shorter distance may be detected in our most powerful telescopes.

Another point of interest in connection with the conception of Lemaître may be the cosmic radiation, coming from all directions. These rays are composed of electric particles, mainly protons, but also α particles and nuclei of heavier elements, moving with speeds of the same order of magnitude as the speed of light. The energies of these particles vary from 2×10^8 electron volts (*e.V.*) up to 10^{18} *e.V.* and even beyond. This is a very amazing fact, for the annihilation of an U atom would only release an energy of about 2×10^{11} *e.V.* In connection with the foregoing the suggestion arises that this radiation may be a relic of radioactive processes in former times, when the atoms were fewer, larger and more explosive. On the other hand there is already evidence that the supernovae emit particles with cosmic energies of same order of magnitude. Recent investigations of the Dutch astronomers Oort and Walraven have shown that the Crab nebula is emitting such cosmic corpuscular rays. Prof. Oort conjectures that these particles have attained their enormous speeds by very quickly varying electromagnetic fields in the former supernova.

Now the initial explosion—in the theory of Lemaître—may be considered as a super supernova and it seems very plausible to assume that also this super super explosion—if it really occurred—was accompanied with an enormous production of cosmic radiation. Thereby we have to bear in mind that the initial explosion took place at any point of the then very small spherical space, so that in later time this radiation must have been still omnipresent, everywhere coming from all directions.

Now it is very probable by the recent investigations mentioned above that

a considerable part of the cosmic rays of higher energy are due to the supernovae, which on average occur once or twice a century in each galaxy. Also there is already strong evidence that cosmic rays of lower energy are emitted by solar eruptions and in similar ways by the stars. But all this evidence does not exclude the possibility that the cosmic rays of the largest energy originate from the super super Lemaître explosion.

5. The Assumption of an Oscillating Universe by Assuming an Inward Directed Force

The latest results of observation—which must still be considered as preliminary—suggest the possibility of a decelerating universe. Therefore let us assume that the system of galaxies collapsed and is now on the rebound, the inward velocities being turned into outward velocities by passage through the centre. Then the velocities are produced by *inward* directed force. Assuming this inward force proportional with the distance, we then obtain:

$$X\frac{d^2R}{dt^2} = -C^2XR \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (17),$$

in which C^2 denotes a constant. From (17) we obtain as solution:

$$R = r\sin\pi\frac{t}{T} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (18)$$

in which R and T are constants and $t = 0$ denotes the time at which the expansion begins (passage through the centre). Then:

$V = X\frac{dR}{dt_1} = \frac{r\pi X}{T}\cos\frac{\pi}{T}t_1$. The time t_1 of light departure corresponding with X then follows from:

$$\int_{t_1}^{t_0}\frac{dt}{\sin\pi\frac{t}{T}} = rX, \text{ or } ln\frac{tg\,\frac{\pi}{2T}t_0}{tg\,\frac{\pi}{2T}t_1} = \frac{r\pi}{T}X, \text{ so that:}$$

$$t_1 = \frac{2T}{\pi}\,\text{arc}\,tg\left(e^{-\frac{r\pi X}{T}}\,tg\,\frac{\pi}{2T}t_0\right)$$

Hence we now obtain as fundamental equations:

$$V = \frac{r\pi X}{T}\cos 2\,\text{arc}\,tg\left(e^{-\frac{r\pi X}{T}}\,tg\,\frac{\pi}{2T}t_0\right) \quad .. \quad .. \quad .. \quad .. \quad (19)$$

$$E^1 = \frac{E}{4\pi r^2\sin^2\frac{\pi}{T}t_0\sin^2 X} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (20)$$

$$\sin X\sin\frac{\delta}{2} = \sin\frac{a}{2r\sin 2\,\text{arc}\,tg\left(e^{-\frac{r\pi X}{T}}\,tg\,\frac{\pi}{2T}t_0\right)} \quad .. \quad .. \quad (21).$$

In (19), (20) and (21) we have X, T, t_0 and r as unknown quantities so that a fourth relation is required. Let us therefore assume that the maximum speed of recession that ever occurs be n (in terms of the speed of light). In connection with (19) we then obtain as fourth relation:

$$\frac{r\pi^2}{T} = n, \text{ or } r = n\frac{T}{\pi^2} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (22)$$

Substituting (22) into (19), (20) and (21), we obtain:

$$V = \frac{nX}{\pi} \cos 2 \text{ arc } tg\left(e^{-\frac{nX}{\pi}} tg \frac{\pi}{2T} t_0\right) \quad .. \quad .. \quad .. \quad .. \quad (23)$$

$$E^1 = \frac{E}{4\frac{n^2T^2}{\pi^3} \sin^2 \frac{\pi}{T} t_0 \sin^2 X} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (24)$$

$$\sin X \sin \frac{\delta}{2} = \sin \frac{a}{\frac{2nT}{\pi^2} \sin 2 \text{ arc } tg\left(e^{-\frac{nX}{\pi}} tg \frac{\pi}{2T} t_0\right)} \quad .. \quad .. \quad (25)$$

In order to solve the remaining unknowns X, T and t_0, let us write:

$$\text{arc } tg\left(e^{-\frac{nX}{\pi}} tg \frac{\pi}{2T} t_0\right) = u. \quad \text{Then by (23):}$$

$$\frac{\pi V}{nX} = \cos 2u, \text{ so that } \sin 2u = \sqrt{1 - \frac{\pi^2V^2}{n^2X^2}}, \text{ and } tg\, u = \sqrt{\frac{1 - \frac{\pi V}{nX}}{1 + \frac{\pi V}{nX}}},$$

whence follows:

$$tg \frac{\pi}{2T} t_0 = e^{\frac{nX}{\pi}} \sqrt{\frac{1 - \frac{\pi V}{nX}}{1 + \frac{\pi V}{nX}}}, \text{ so that:}$$

$$\sin \frac{\pi}{T} t_0 = \frac{2\sqrt{1 - \frac{\pi^2V^2}{n^2X^2}}}{\left(1 + \frac{\pi V}{nX}\right) e^{-\frac{nX}{\pi}} + \left(1 - \frac{\pi V}{nX}\right) e^{\frac{nX}{\pi}}} \quad .. \quad .. \quad (26)$$

From (24) and (26) it follows:

$$\frac{2nT}{\pi^2} = \frac{\left(1 - \frac{\pi V}{nX}\right) e^{-\frac{nX}{\pi}} + \left(1 - \frac{\pi V}{nX}\right) e^{\frac{nX}{\pi}}}{2\sqrt{1 - \frac{\pi^2V^2}{n^2X^2}} \cdot \sin X} \sqrt{\frac{E}{\pi E^1}} \quad .. \quad .. \quad (27)$$

From (25), (27) and the relation $\sin 2u = \sqrt{1 - \frac{\pi^2 V^2}{n^2 X^2}}$, we then find:

$$\sin X \sin \frac{\delta}{2} = \sin \frac{2a \sin X}{\left[\left(1 + \frac{\pi V}{nX}\right) e^{-\frac{nX}{\pi}} + \left(1 - \frac{\pi V}{nX}\right) e^{\frac{nX}{\pi}}\right] \sqrt{\frac{E}{\pi E^1}}} \qquad (28)$$

From (28), X can be found numerically. Then T follows from (27) and consequently t_0 from (26). This procedure must be tried for different values of n. If for different groups of nebulae (corresponding with different values of V) same values for T and t_0 are found, this will be a strong indication that the supposition (18), with the corresponding specially assumed value for n, is in accordance with the truth.

Further the geodesic distance corresponding with X at the corresponding time of light emission becomes:

$$XR_{t_1} = rX \sin 2 \operatorname{arc\,tg}\left(e^{-\frac{r\pi X}{T}} \operatorname{tg} \frac{\pi}{2T} t_0\right) \quad .. \quad .. \quad .. \quad .. \qquad (29)$$

Then we find by (19) and (29):

$$\frac{V}{XR_{t_1}} = \frac{\pi}{T} \operatorname{cotg} g\, 2 \operatorname{arc\,tg}\left(e^{-\frac{r\pi X}{T}} \operatorname{tg} \frac{\pi}{2T} t_0\right) \quad .. \quad .. \quad .. \qquad (30)$$

Now because in first approximation the expansion from the beginning up to the present seems to be linear with time, we have to assume:

$$t_0 \ll T.$$

Then as first approximation we may write (29):

$$XR_{t_1} = 2rXe^{-\frac{r\pi X}{T}} \operatorname{tg} \frac{\pi}{2T} t_0 \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \qquad (31)$$

so that:

$$\frac{d}{dX}(XR_{t_1})\, dX = 2r \operatorname{tg} \frac{\pi}{2T} t_0 \left(1 - \frac{r\pi X}{T}\right) e^{-\frac{r\pi X}{T}} dX \quad .. \quad .. \qquad (32)$$

Hence if $\frac{r\pi X}{T} > 1$ (33), XR_{t_1} increases with decreasing X.

Now the inequality (33) can only occur if $\frac{r\pi^2}{T} = n > 1$, hence if the maximum speed of recession which ever occurs is larger than the speed of light. Further, according to (30) also the ratio V/XR_{t_1} will decrease with decreasing X, that is with increasing XR_{t_1} if (33) is available. Then at very large distances (always corresponding with the time of light departure of the light now received) the speeds of recession will be smaller than they should be if the speed of recession were still proportional to the distance. The latest preliminary results of observation suggest that this feature is in accordance with truth.

6. On the justification of the hypothesis of spherical space time

In the last model (harmonically oscillating universe) the radius of curvature varies with time according to the formula—

$$R = r \sin \boldsymbol{T} \quad (\boldsymbol{T} = \pi \frac{t}{T}, \text{ compare with (18)})$$

in which $\boldsymbol{T}$ is now expressed in arc measure. If $\boldsymbol{T}$ varies from 0 to $\frac{\pi}{2}$, R varies from 0 to r, whereas from $\boldsymbol{T} = \frac{\pi}{2} \rightarrow \boldsymbol{T} = \pi$, R again decreases to 0. Hence one oscillation is passed through if $\boldsymbol{T}$ varies from 0 to π. The history of one oscillation of this universe can be represented by a hyperspherical four dimensional continuum with squared line element:

$$ds^2 = r^2[d\boldsymbol{T}^2 + \sin^2 \boldsymbol{T}\{dX^2 + \sin^2 X(d\theta^2 + \sin^2 \theta d\phi^2)\}].$$

The time $\boldsymbol{T}$ is "cosmical time"; the cosmical time direction in any small and therefore flat region of hyperspherical space time is prescribed; it is the direction of the worldline of a point with respect to which the distribution of the speeds of recession of the spiral nebulae is radial symmetric. The centre of the Milky Way system can practically be considered as such a point, corresponding with cosmical time. (Thereby we assume that the speeds of the spiral nebulae are only due to the expansion of space; small superimposed motions are disregarded.) $\boldsymbol{T} = 0$ denotes "the beginning of expansion" and corresponds in hyperspherical space time with the radiant of worldlines of spiral nebulae.

Now the constituting elements of the fundamental physical world are not instantaneous objects, but *events*. Hence we have to consider not the oscillating spatial universe, but hyperspherical space time as ultimate physical reality, only corresponding with one oscillation. [The repetition of the same stage of universe, in which *all* is included, can in principle never be observed and has therefore no meaning.] In this spherical space time, $\boldsymbol{T}$ *always* denotes cosmical time; for the rest the choice of the space co-ordinates X, θ and ϕ is not prescribed.

The idea that the world is not only curved in space but also in time and that space-time must be spherical reconciles with the hypothesis that "everywhere in space-time" the general character of space-time is the same. This view is quite evident as soon as we accept the outlook that not the instantaneous objects, but the events themselves are the constituents of reality. Also the property of spherical space-time, that the geometry is not invariant as to conformal transformations, is in accordance with the fact that the physical properties of material systems vary as well as to such transformations.

Further, the interesting question arises that even *many* radiants of galactic worldlines may exist. For if we observe one radiant in the part of the world directly accessible to us, the supposition seems natural that this observed radiant will not be unique. If then R should appear to increase proportionately with $\boldsymbol{T}$ since $\boldsymbol{T} = 0$, this should mean in our model that our situation in hyperspherical space-time should be in the neighbourhood of the radiant $\boldsymbol{T} = 0$. For if we denote our present by $\boldsymbol{T} = \epsilon$, then only for small values of ϵ, $r \sin \boldsymbol{T} \approx r\boldsymbol{T}$, in the interval $\boldsymbol{T} = 0 \rightarrow \boldsymbol{T} = \epsilon$.

Speaking in terms of space and time, we may consider the expansion from $\boldsymbol{T} = 0$ as a process of dissipation of energy, so that the entropy gradient in the

neighbourhood of a radiant may have a radially outward direction with respect to this radiant, and may correspond with the *arrow of time*, in which "time" is cosmical time. Hence the arrow of time should only have local significance in the four dimensional world. Somewhere between the galactic radiants, there should be no arrow of time at all.

Further it will be evident that the conception of a harmonically oscillating universe—as outlook from the view point of separated space and time—is no more the counterpart of a spherical four dimensional substratum populated with radiants of galactic worldlines. This harmonically oscillating universe should only be this counterpart, if the structure of the visible universe (in space and time) could be extrapolated over the whole in physical sense. If this extrapolation in physical sense should be really in accordance with truth, we should only have two radiants, mutually opposite, and these two radiants could then further be combined into one by representing space-time no more as hyperspherical substratum, but as a so-called *elliptical continuum*. This elliptical continuum is then a slightly modified kind of spherical continuum which we obtain if the opposite points of the spherical continuum are considered as identical (indistinguishable). It does not seem probable that such a model (four-dimensional elliptical space-time) will be an adequate overall representation of the whole—although this possibility cannot be excluded.

REFERENCES

(1) Eddington, *The Expanding Universe*, p. 35. (2) *Ibid.*, p. 25. (3) *Ibid.*, p. 26. (4) *Ibid.*, pp. 23, 49. (5) *Ibid.*, p. 78.

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FIG. 1.

A warhead instrumentation section of a V-2 rocket is being hoisted by the crane of the mobile gantry. This shows how the V-2 nose section was considerably modified for upper atmosphere research.

HIGH ALTITUDE RESEARCH

By Eric Burgess, F.R.A.S.

(*A paper read to the British Interplanetary Society in London on December* 3, 1955)

SUMMARY

A brief review is given of the development of vehicles for the exploration of the upper atmosphere, together with the peculiarities connected with rocket experiments in this field. Types of experiments undertaken and brief particulars of the instrumentations involved are described. Methods of data recovery are explained and it is suggested that following the successful launching of satellite vehicles, rocket probe missiles might be sent to the Moon and to other planetary bodies of the Solar System.

Before man can hope to achieve interplanetary missions and to establish permanent satellites he must have more accurate knowledge concerning the structure and composition of the terrestrial atmosphere at extreme altitudes. Prior to World War II the exploration of the upper atmosphere was undertaken indirectly by the observation of natural phenomena such as the aurorae, meteors, and noctilucent clouds, and also by the use of probes such as radio waves.[1]

At the close of hostilities data became available on the V-2 and it was realized that the long-range rocket could be capable of carrying instruments to altitudes far greater than those which had previously been attained. Yet even earlier many scientists had been thinking of using rocket vehicles for the exploration of the atmosphere to levels higher than those reached by sounding balloons. In fact these thoughts had led to the design of the WAC Corporal,[2] a liquid propellant rocket missile which was boosted by a solid-propellant rocket, and was launched from a tower to reach an altitude of 43 miles.

However, before the WAC Corporal could be put to extensive use in this field the war ended and a large number of German V-2 rockets were captured and transported to the U.S.A., where they were to be fired by the United States Army in order to evaluate the weapon and to give Army personnel experience in the handling of these vehicles. The stability of the V-2 depended upon it carrying a warhead payload of approximately 1 ton and it seemed rather a pity to carry this weight of concrete or other ballast a hundred miles into the upper atmosphere. Accordingly, the warhead weight and space were made available for scientific instruments (Fig. 1) and an upper atmosphere rocket research panel was formed which allocated the space in the various firings to research agencies and universities.

In all, 66 V-2's were launched over a period of six years and the greatest altitude reached was 133 miles. A typical V-2 instrumentation would include such instruments as temperature and pressure sensitive devices, a cosmic-ray telescope, a spectrograph, and cine-cameras for photographing the surface of the Earth. In addition there might be a telemetering transmitter to return the data to the ground stations in the form of a coded radio broadcast, a doppler transreceiver for measuring the velocity of the rocket, a cut-off receiver for range safety, and transmitters which would be used to determine the electron density in the ionosphere.

While the V-2 rockets were being used the WAC Corporal was developed to produce a new high-altitude rocket known as the Aerobee.[4] The vehicle is 26 ft. long and 15 in. in diameter and carries a payload of 120 to 300 lb. It uses a liquid-propellant sustaining motor and is launched with the aid of a solid-propellant booster the size of which can be varied to send the vehicle to various altitude levels.

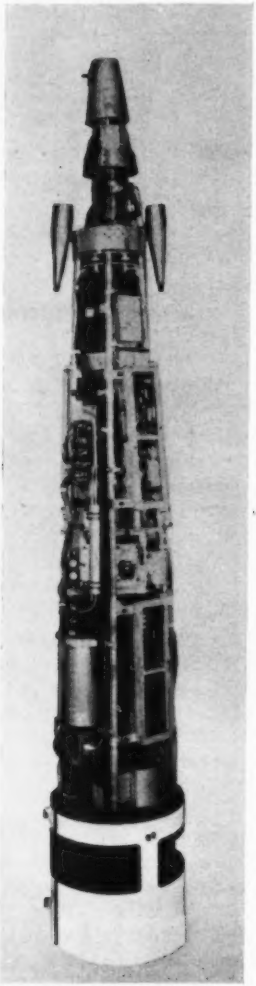

FIG. 2. A typical instrumentation of an Aerobee nose cone by Cook Research Laboratories.

Launching is effected from a special tower, which is 142 ft. high. There are no methods of directing or controlling the rocket after it has left the tower, so this is given a permanent tilt at Holloman Air Force base 1 degree towards the north in order to give a directional aim to the rocket. In addition the tower can be tilted through an angle of 2 degrees to the east or 1 degree to the west, so that the rotation of the Earth and the effects of winds can be compensated for and the rocket kept within the proving area.

The sustaining motor and booster are fired simultaneously. The booster burns for only a few seconds to give a peak acceleration of 14 g to the 1,100-lb. rocket. The sustaining motor burns for 33 sec. and burn-out occurs at 30 miles altitude, when the rocket is travelling at some 3,600 m.p.h.

On the downward leg of the trajectory the streamlined tail fins are blown off at 200,000 ft. so that the missile becomes aerodynamically unstable and begins to tumble. At 20,000 ft. the nose-cone containing the instruments (Fig. 2) is blown off to complete the remainder of its journey by parachute.

A large number of Aerobee rockets have been fired and many more are planned for the International Geophysical Year. It is a highly successful rocket and has been currently modified to produce the Aerobee-Hi, which attained altitudes of over 150 miles in 1955 (Fig. 3).

A larger high-altitude rocket was also required to replace the V-2 and this demand produced the Glenn L. Martin "Viking," which could carry a payload similar to the V-2, but to greater altitudes. A number of "Viking" rockets have been fired successfully, the present altitude record being 156 miles.[5] Project Vanguard stems from work done in the development of the "Viking."

There are a number of problems peculiar to high-altitude research with rocket vehicles. When the rocket motor ceases to thrust the missile itself continues to ascend under the influence of

FIG. 3. One of the most modern and successful high-altitude rockets is the Aerobee-Hi, which has reached altitudes of over 150 miles. This photograph shows an early version of the missile being placed in its launching tower prior to firing.

momentum and, in fact, the greater portion of its flight is along a free trajectory. If there is any residual angular momentum at the instant of motor cut-off the rocket may pitch, yaw, or roll, and will continue to do so during the rest of its flight until it re-enters the appreciable atmosphere. Such random motion can invalidate experiments and accordingly attempts have been made to control the orientation of the vehicle once the motor has ceased firing. While thrust is being developed the rocket can be orientated by deflecting the jet, either by

FIG. 4. Orientation of the rocket vehicle during free fall can be checked by means of a special camera which produces wide-angle photographs of the Earth. One of these photographs taken from a Viking at 108 miles is shown here. Orientation is determined from the position of landmarks and the horizon.

FIG. 5. Recovery of instruments is possible from the debris of the high altitude missile especially if air break-up is achieved by destroying the aerodynamic characteristics of the vehicle. The photograph shows how some parts of the vehicle are recovered almost intact.

FIG. 6. Recovery of instruments from the fallen parts of the rocket calls for an efficient recovery service and this is achieved by the use of helicopters and land vehicles acting on instructions from the radar plotting section at base headquarters. The helicopter makes the preliminary search to locate the impact point and then the surface vehicles bring along the search parties to retrieve the parts which may be scattered over a wide area.

means of vanes dipping into it or else by mounting the motor on gimbals. After burn-out several methods are still possible, for example, to maintain the orientation of the whole or just a part of the rocket vehicle. For the whole rocket a jet system is needed and in the case of the Viking steam jets were used to preserve a constant heading. The alternative is to use an instrument similar to a sun-seeker[6] whereby a part of the rocket is mounted free to rotate about two axes and is controlled by photo-electric devices to maintain constant orientation with respect to the sun. The sun-seeker device has been used in practice to obtain long-exposure photographs of the solar spectrum in regions where the intensity of radiation is low.

Accurate knowledge of the orientation is also most important in order that the data gathered by the instruments within the rocket can be correlated with the trajectory data. A system of gyroscopes might be used to do this; alternatively, as in certain Viking experiments, an orientation camera[7] having a wide-angle lense is used to photograph the Earth. From the positions of landmarks or the image of the sun and that of the Earth's horizon, it is possible to determine the orientation of the rocket at each instant during its flight (Fig. 4).

The next problem in rocket exploration is concerned with the recovery of data, and this is also of prime importance when extra-terrestrial probe missiles are contemplated. A high altitude rocket which reaches an all-burnt velocity

FIG. 7. Recovery of data is also possible by the use of ribbon parachutes which bring the nose cone of the rocket to Earth at a low impact velocity. This photograph shows the recovery of the nose-cone of an Aerobee rocket which was instrumented for gathering samples of the upper atmosphere. The scoops are visible projecting to the left of the missile body.

of over 3,000 m.p.h.—similar to the V-2 or the Viking—will also re-enter the dense regions of the atmosphere at that speed. If it is a streamlined body, such as the V-2 in nose-down attitude because of the effect of its tail fins, it will penetrate the atmosphere and hit the surface at a high velocity. The result is that a deep pit is dug in the floor of the proving ground. A crater from a V-2 impact, for example, may be as much as 50 ft. deep with the desert disturbed to a depth of over 80 ft. From a crater like this it is virtually impossible to recover successfully any instruments and their records.

Again there are several ways of ensuring that data are recovered from the rocket vehicles. Early experiments with V-2 rockets employed the ejection of a small compartment which had a poor aerodynamic shape and would accordingly fall at a fairly low velocity. Impact would be at about 200 to 300 m.p.h., but the serious difficulty was the location of the small object amongst the desert vegetation.

The next method was that of breaking up the missile in flight so that the parts would have high drag and low impact velocity. A large number of rocket vehicles were recovered in this way (Fig. 5) either by blowing off the nose cone or the tail skirt, and both the afterbody and the warhead were usually recovered in good condition by the search teams (Fig. 6). Although this method of

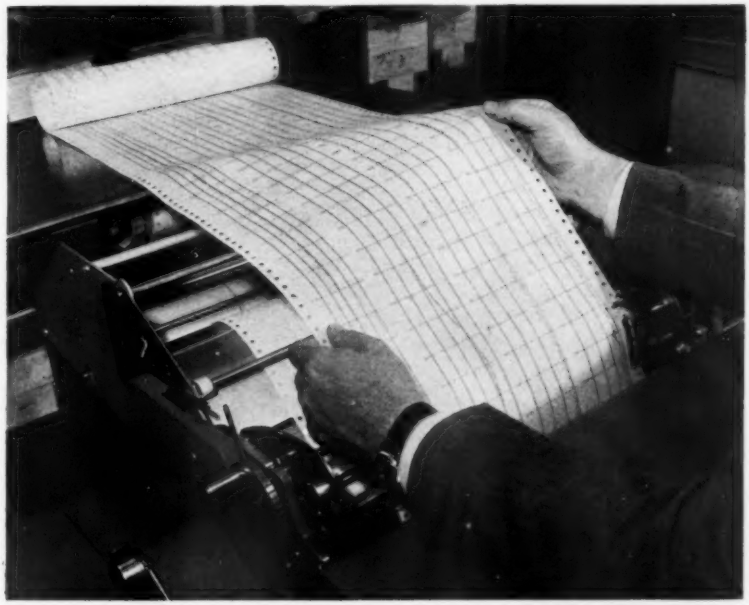

FIG. 8. Data telemetered from high altitude missiles are recorded by means of photography of cathode ray traces, by magnetic tape and also by pen recorders as shown in this photograph. Plotting machines draw hundreds of graphs representing what has happened during the few minutes of a single missile flight.

recovery sufficed for a number of experiments, others required a still lower velocity of impact, for example, aero-medical experiments with mice and monkeys. In consequence attempts were made to recover parts of the rocket vehicle by a parachute system. This was found to be quite possible in the case of Aerobee nose cones, which were recovered from altitudes of 60 miles (Fig. 7), but difficulties were encountered with larger vehicles falling from above the 100-mile level.

Perhaps the most useful method of data recovery is the telemetry system, which consists of a coded radio signal on which data is impressed by some kind of modulation.[8] An early airborne unit employed in the V-2 experiments relied upon the instruments converting their readings to voltages which were passed through a calibrator to a series of multi-vibrators which were selected in turn to pass the data to the transmitter. Pulses from this transmitter were arranged to follow in sequence, the time between them being a measure of the magnitude of the data voltages. The ground station, after receiving the signals, decoded them to make it possible for the original data voltages to be recorded by means of cine film, magnetic tape and in the form of graphs drawn by pen recorders[9] (Fig. 8). Some interesting side results were obtained from early experiments which showed how the received signal strength was affected by the ionized

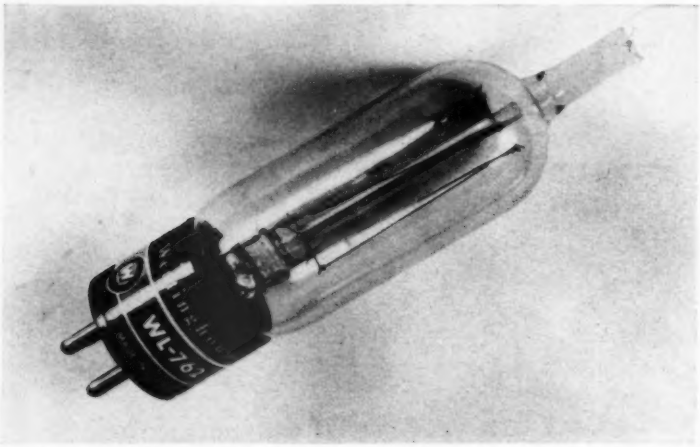

FIG. 9. Pressures in the upper atmosphere are measured by several different types of gauges. A Pirani gauge of the type shown here was mounted near the tail of the rocket—just forward of the tail fins—where it was found from wind tunnel data that the pressure on the V-2 skin would be similar to ambient pressure.

layers of the atmosphere and by ionization in the exhaust jet of the rocket itself.

The first series of experiments made with high-altitude rockets was concerned with the physical conditions of the upper atmosphere. Atmospheric pressure was determined as a function of altitude by using a number of different types of pressure sensitive gauges (Fig. 9). These included pirani gauges, ionization gauges and alphatrons, while a simple bellows gauge was used for the lowest regions. From the difference in pressure between the tips and the walls of cones mounted at the front of the rocket vehicles it was possible to determine the Mach number of the vehicle. Knowing the rocket's velocity it was then possible to establish the velocity of sound as a function of altitude and thus determine the atmospheric temperature. Temperatures were thus obtained indirectly from the pressure measurements because any temperature sensitive device would be moving too quickly through the atmospheric layers for it to give accurate readings. Moreover the skin temperature of the missile itself is governed mainly by aerodynamic factors and even neglecting these, then solely by the balance between radiation received and emitted. Measuring the skin temperature of a stationary body would not give ambient temperature.

It was found that the results obtained with the rocket-borne instruments differed somewhat from those calculated from ground-based observations. The rocket temperatures on the whole were found to be lower than those previously calculated, but the form of the curves of temperature as a function of altitude were very much the same; the temperature maximum and minimum regions coincided on the two sets of results.

One pressure instrumentation used a number of ionization gauges which were sealed so that they would not be damaged by exposure to high density air

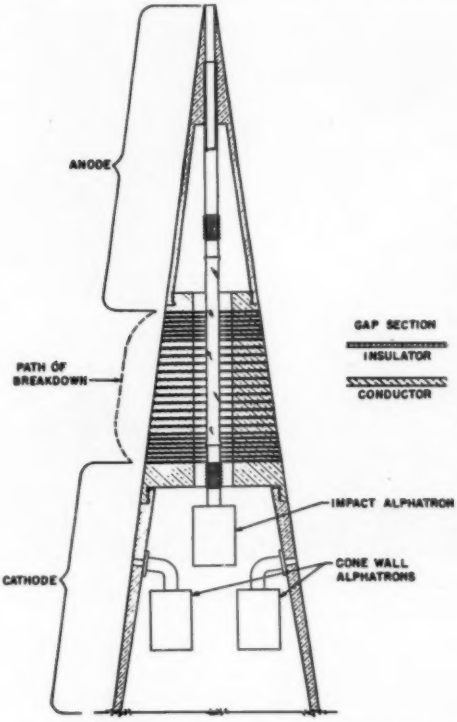

FIG. 10. Another method of measuring upper atmospheric pressures is by using alphatrons which are really ionization gauges using a radio-active substance to produce ionization. By comparisons of cone wall and ram pressures it is possible to calculate the ambient temperature. Also in the instrumentation shown diagrammatically here an attempt was made to determine the atmospheric characteristics by means of a voltage breakdown technique. The experiment worked successfully, but the results were not reliable because the characteristics were measured just behind the nose shock.

at low altitudes. A timing device allowed a spring mechanism to throw off a protective hollow cone. This broke glass tubulations and opened the gauges to the atmosphere.[10]

Attempts were made also to measure the density of the upper atmosphere by a voltage breakdown method (Fig. 10), but although the experiment worked successfully, the results could not be regarded as absolute ones because the measurements were necessarily made behind the nose shock wave.

Another important series of experiments with high-altitude rockets is that concerning the ionized layers. From the Earth's surface these layers are investigated by radio methods which rely upon a series of pulses of gradually increasing frequency being transmitted vertically. The time between the emission of the pulse and the reception of its echo is measured,[11] and the time delay can be used to calculate the altitude of the refracting layer which returns the pulse to the Earth's surface. As the frequency increases there is suddenly an increase in the time delay which indicates that the radio waves are then passing through the lower ionized layer and being returned from a higher one, which is more strongly ionized.

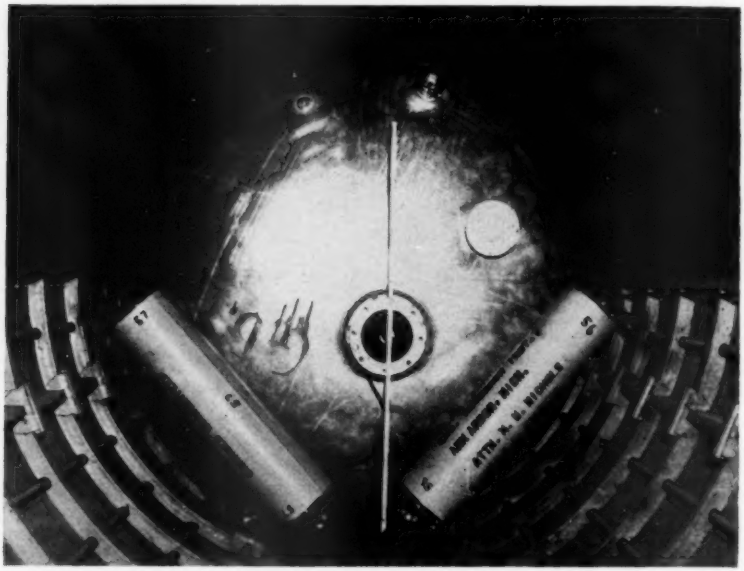

FIG. 11. Attempts have been made to check diffusive separation of the Earth's atmosphere by carrying sampling bottles in rockets. Here two such bottles are seen mounted in a V-2 rocket just forward of the alcohol tank.

The shortcoming of this technique is that although the density of ionization can be calculated for the various layers, no information can be gathered concerning the region between the maximum ionization in one layer and the same ionization in the next higher layer. This region can only be investigated by carrying instruments aloft in rocket vehicles.

At an early stage in the upper atmosphere research with V-2 rockets ionosphere experiments were undertaken.[12] These consisted of using a transmitter which could radiate a radio frequency and one of its harmonics. A probing frequency would be chosen which was close to the critical frequency at the time of the rocket experiment. There would be a phase retardation as the radio waves passed through the ionized layers. The harmonic was sufficiently high in frequency for it to pass through the layer without any phase retardation so that it could be used as a reference frequency. At the ground station the lower frequency signal was applied to a frequency multiplier and a beat was produced between the resultant and the control frequency. From the frequency of the beat it was possible to determine the effective electron density in the regions through which the rocket had passed.

The experiment presented a number of difficulties. First because it relied upon the production of beats it demanded that there should be no spurious phase shifts arising in the apparatus itself. Secondly the provision of sufficiently long aerials for the radiation of the low frequency was difficult. It was accom-

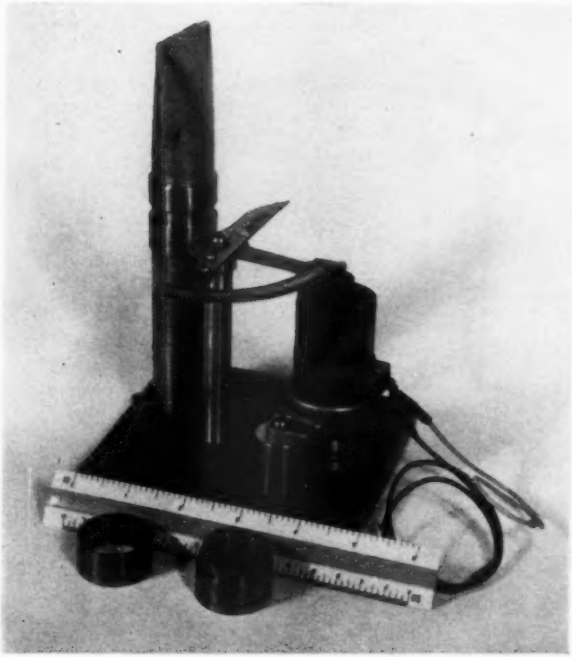

FIG. 12. A mechanical device is used to open the bottles and allow air to pass into them. Afterwards an explosive device (not shown here) seals the inlet tube.

plished by using delta aerials mounted between the mid-section and the tail fins. During the ensuing years much more refined experiments were devised and conducted and a large amount of data concerning the state of the ionosphere has been accumulated.

Further experiments were concerned with the composition of the atmosphere itself and if diffusive separation of the atmospheric gases takes place. This was tackled in two ways, first by the inclusion of sampling bottles within the rockets (Fig. 11), bottles which could be opened at the peak of the trajectory (Fig. 12) then sealed again so that the contents could later be chemically analysed after recovery of the rocket.[13] A sampling bottle technique has to be very carefully arranged so that the gases exuding from the rocket structure are not taken into the sampling bottle. One method of doing this is to mount the scoops at the front of the rocket vehicle and to blow off the structure in front of the scoops by means of a JATO bottle. In practice it was found that these sampling techniques produced conflicting results so that it was not determined from them whether or not diffusive separation was taking place.

Another attempt to solve the problem was made by two agencies using mass spectrometers carried in V-2 and Aerobee rockets[14] (Fig. 13). The first experiments were not successful because of rocket failures, but later tests showed that there was no diffusive separation at least up to an altitude of 100 miles.

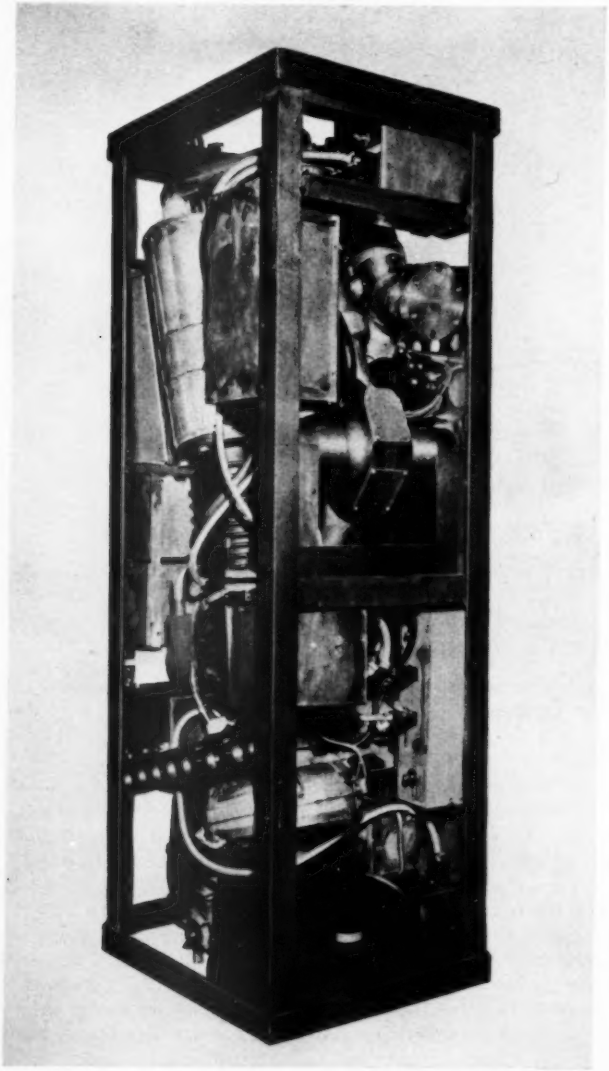

FIG. 13. Another method of checking the constituents of the upper atmosphere is by the use of a radio-frequency mass spectrometer mounted in the rocket and able to telemeter its data. This photograph shows an instrument of this nature which was developed by Cook Research Laboratories and flown in V-2 rockets. It measures the relative abundance of helium and argon at the different levels of the atmosphere.

Attempts were also made to detect the presence of micro-meteorites by the use of polished plates and specially sensitive microphones mounted on the skin of the missile. Again although there were indications that a number of small bodies had impinged on the plates and the microphones, there seemed to be no altitude dependence. The results were not conclusive as to whether or not the observed impacts represented micro-meteorites or merely particles present in the terrestrial atmosphere and arising from terrestrial sources.

Perhaps the greatest use of the rocket vehicles has been in connection with solar physics. Not only does this research give valuable information to the astronomer, but it is also of great use to the geophysicist who is interested in the photo-chemical processes taking place in the upper terrestrial atmosphere. Although we may think when we look up into a clear night-sky that we are fortunate in being born on a planet with a transparent atmosphere which permits us to see the surrounding universe, in fact our atmosphere is only transparent to small ranges of wavelengths of the complete electro-magnetic spectrum. To many wavelengths the atmosphere acts as an impenetrable barrier.[18]

There are in fact two windows; the optical window extending from about 3,000 A to 30,000 A, and the radio window between approximately 1 cm. and 30 m. If we are content to observe the universe by means of radiation at frequencies within these limits all is well. But very important regions of solar and stellar emission lie beyond them and the radiation cannot penetrate to observatories at the Earth's surface. A most important region of the solar spectrum extends into the extreme ultra-violet from 3,000 A and includes radiation which is responsible for a number of photo-chemical and ionization processes in the upper atmosphere. Consequently when it was realized that rocket vehicles could carry instruments above the main absorbing layers attempts were made to instrument these rockets with spectrographs and photo-electric devices which could investigate the solar spectrum even as far as the X-ray region.

Preliminary calculations showed that above an altitude of 15 miles the solar spectrum could be extended to just below 2,000 A and a spectrograph was designed capable of photographing the solar spectrum into this region (Fig. 14). It consisted essentially of two lithium fluoride beads to give point images of the sun at a wide range of orientations of the rocket, and a concave diffraction grating which dispersed the spectra on to ultra-violet sensitized 35-mm. film at the base of the instrument. The film was moved by a pre-set timing mechanism which finally wound it into a casette of armour-piercing steel so that it could be stored safely to survive the impact and later be recovered for processing.

In the first experiments[16] the spectrograph was mounted at the tip of the warhead, but recovery was found to be difficult in this position so that later models were flown in one of the fins (Fig. 15). A successful series of spectra was first obtained from the firing of October 10, 1946. At 17 km. altitude the solar spectrum cut off abruptly at 3,000 A, whereas the highest successful spectrum obtained in the 55-km. region extended to 2,350 A.

The main absorber of solar ultra-violet is the layer of ozone at an altitude of approximately 20 km. above sea level. By means of a series of spectra it is

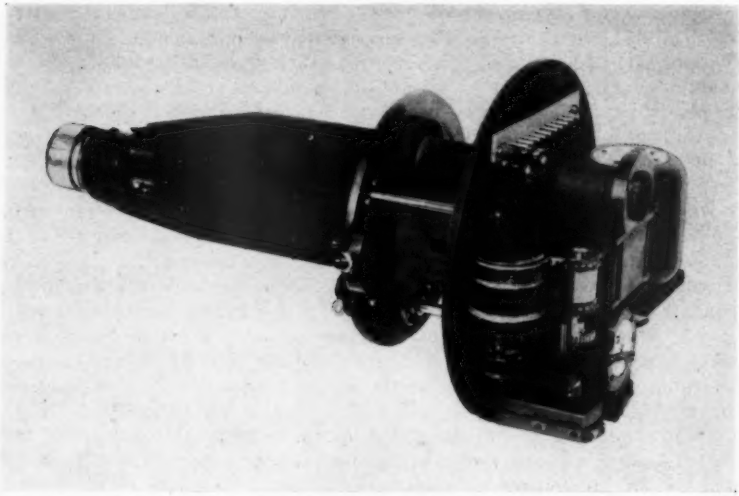

FIG. 14. A side view of the ultra-violet solar spectrograph flown in V-2 rockets. The film reservoir, sprocket and storage casette are visible. On the mid bulkhead is the bead mount and the shutter housing.

possible to calculate the vertical ozone distribution above the launching site. It is found that the curves obtained agree quite well with the ground-based observations of ozone concentration.

In addition the rocket experiments have enabled a large number of absorption lines in the solar ultra-violet spectrum to be catalogued, interesting lines being the Si I, Mg I and the Mg II doublet. Attempts have also been made to trace the solar spectrum photographically into the extreme-violet. It has indeed been recorded to 1,900 A while one photograph extends even to 1,700 A. It is found that there is a rapid falling off in intensity beyond 2,100 A which could be due to some absorbing process in either the atmosphere of the Earth or that of the Sun.

In order to investigate the solar spectrum in the far ultra-violet regions where the intensity is low it is necessary to employ some kind of sun-following device in order to keep the maximum amount of illumination entering the apparatus despite any rolling or tumbling of the rocket. This has been accomplished by means of sun-seekers. One arrangement used a system of mirrors whereas a more advanced design employed a device whereby most of the nose of an Aerobee incorporated a spectrograph which could be moved about two axes. It was kept pointing towards the Sun by means of photo-electric detectors operating servomechanisms.

An outcome of the research in solar physics has been the determination of the intensity of spectral irradiance of the Sun into the far ultra-violet showing that the Sun emits radiation similar to a black body at a temperature of only 4,500° K at a wavelength of 2,000 A compared with nearly 6,000° K in the visible

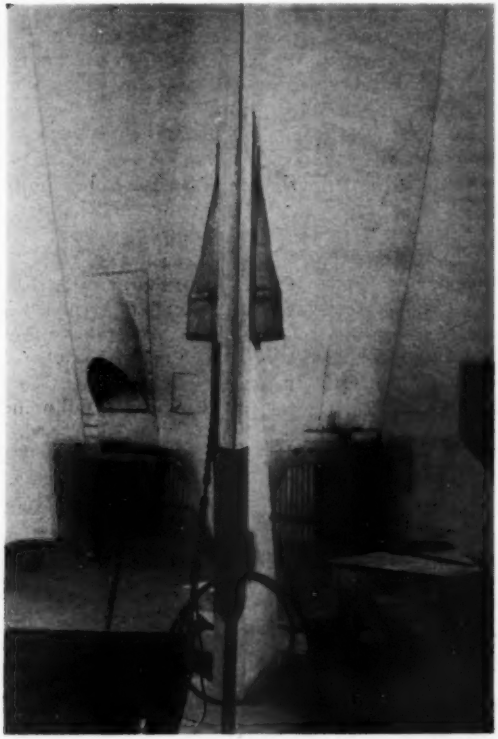

FIG. 15. Although early spectrographs were mounted at the nose of the V-2 it was found that this was not a satisfactory place for recovery. Accordingly the first successes were obtained when the spectrograph was mounted in one of the fins of the rocket as shown in this photograph.

region. It has also been possible to revaluate the solar constant at a figure of 2·00 ± 0·04 calories per sq. cm. per min. The Lyman alpha emission line at 1,216 A has also been detected photo-electrically.[17]

High altitude rockets have made it possible to investigate the cosmic-ray primaries beyond the appreciable atmosphere. Early experiments using fairly simple assemblies of counter tube checked cosmic ray intensity as a function of altitude. It was found that decreasing atmospheric pressure gave rise to increasing counts of cosmic-ray particles. A maximum was reached at a pressure of just under 100 mm. of mercury, but the counting rate decreased as the pressure fell further with increasing altitude. The maximum occurs when the incoming cosmic-ray primaries have reacted with the atmospheric gases and produced a large number of secondary and other particles which have not had time to enter into absorbing collisions.

More refined cosmic ray experiments have been made both with single counters and more complex instrumentations such as cloud chambers. One of the latter carried aloft in a V-2 rocket to an altitude of 140 km. in January, 1948, produced the first photograph of a cosmic-ray primary beyond the

appreciable atmosphere. From many experiments it has now become possible to plot a smooth curve of counts per sec. as a function of altitude for the geomagnetic latitude of 41° N. This curve shows a smooth rise to maximum counts at an altitude of 20 km. The counting rate then decreases with increasing altitude until it reaches half the maximum rate at an altitude of 50 km. Beyond that height the rate remains constant at least to an altitude of nearly 200 km.

Rocket research has thus considerably extended our knowledge of the upper atmosphere and made it important that missiles should now start probing into interplanetary space. In many cases the research has confirmed what was already known while in some matters it has raised new problems.

The state of our knowledge can be summarized as follows:—

Beyond the troposphere we find the first temperature minimum in the stratosphere and modern rocket planes ascend into this region and to the level where cosmic-ray particles reach the maximum counting rate. Just above this height is the ozone layer where incoming solar radiations convert the atmospheric oxygen into ozone. The absorption of energy during this process leads to an increase in atmospheric temperature so that there is a temperature maximum layer lying above the ozone layer at an altitude of between 50 and 60 km. Moreover up to this height both molecular oxygen and molecular nitrogen are present.

Just above the temperature maximum is the region where the lowest aurorae appear and it has been probed by rockets like the WAC Corporal.

A little higher, at 80 km., there is the second temperature minimum layer, a region in which the noctilucent clouds appear, and higher still we come to the lowest ionized layer, the D layer, which is very susceptible to solar ultra-violet. Above the D layer the oxygen molecules begin to dissociate under the action of solar radiation into atomic oxygen.

At 100 km. is the E layer in which electrojets flow and which were detected by early Aerobee rockets penetrating into this region. About an altitude of 140 km. the atmosphere consists principally of molecular nitrogen and atomic oxygen while just below 200 km. the solar radiation is sufficiently intense to break down the nitrogen molecule and produce atomic nitrogen. The V-2 rocket has penetrated almost to this region, while the Viking and Aerobee-Hi ascend even higher into the F1 region at about 250 to 300 km. The highest ionized layer, the F2, is at about 400 km. and has been penetrated by the Bumper-WAC.

The atmosphere consists of a certain amount of ionized nitrogen molecules, atomic nitrogen and atomic oxygen with traces of helium and hydrogen at 700 km. The mean free paths of the atmospheric gases are so long that the particles follow dynamic orbits under the influence of the Earth's gravitational field. We are in the region which is known as the exosphere. But the atmosphere extends even higher than this, the highest sunlit aurorae being observed at 1,200 km. Beyond that altitude we can say that we would most probably be in interplanetary space where the interplanetary gas consists mainly of ionized hydrogen and free electrons together with appreciable quantities of meteoritic dust.

As has been pointed out elsewhere the next stage in upper atmosphere and

solar physics using rocket vehicles is going to be the establishment of satellite vehicles, first in temporary orbits and then in permanent ones. For permanancy the orbit must lie above a certain minimum height so that air resistance may be neglected. The temporary satellites will be used to determine this minimum height.

The method of entry into a satellite orbit can take two extreme forms; one is economical in time while the other is economical in energy requirements.[18] The first, time-saving, method is to project the vehicle vertically with sufficient energy for it to ascend to the altitude of the chosen circular orbit. When it reaches that altitude it would commence to fall back along the radius vector to the planet's surface, but at this point it is given a further impulse which accelerates it horizontally to the velocity needed in the circular orbit. This is a very wasteful procedure as far as energy requirements are concerned for any orbits which lie at any appreciable distance above the surface of the Earth.

The alternative extreme is the most economical in propellants, but takes the longest time and demands perhaps finer control instrumentation. The missile is accelerated to the perigee velocity in an ellipse which comes close to the surface at perigee and rises to the level of the circular orbit at apogee. The missile coasts half-way around the planet along this ellipse and a further impulse is given at apogee to change the elliptical velocity to circular velocity. Even for this optimum elliptical entry orbit it is found that certain orbits need a characteristic velocity exceeding escape velocity for their establishment. This arises because it is necessary to carry propellants through the gravitational potential of the Earth for the second application of thrust.

The artificial satellite now has extensive references in the literature, but one naturally enquires as to what is likely to be the next stage following the establishment of Earth satellites. This is hardly likely to be manned flight into space. The next immediate step will undoubtedly be some attempt at sending guided missiles to explore the Moon, either to circumnavigate the Moon and obtain photographic records of the hemisphere which is perpetually turned away from the Earth or alternatively to land on the Moon and to telemeter data back to this planet.

Either of these two missions could be attempted with vehicles not much larger than that needed for the establishment of a permanent Earth satellite. We might indeed send a missile to the Moon which could land on the lunar surface with a specially developed seismograph and then later fire other missiles which would land on the Moon and explode atomic warheads. By checking the explosion waves through the lunar material the internal structure of our satellite might be determined. The missile could at the same time check if the grey areas of the lunar surface (Fig. 16) really do consist of deep layers of dust as has been suggested by some astronomers.

Afterwards it is most likely that attempts will be made to send probe rockets to other planets. Velocities only slightly in excess of escape velocity are needed to carry probe missiles deep into interplanetary space. Mercury, Venus, Mars, Jupiter are all within the reach of chemically-fueled probe missiles.

It is most likely that Mars will be chosen as the target for the first interplanetary missile and a few calculations will show that for a velocity only

FIG. 16. Eroded area of the Moon in the region of the Ariadaeus cleft. A missile landing on this area facing the Earth would be in the best position for telemetering data concerning lunar surface conditions.

0·2 km./sec. faster than escape velocity a missile can travel from the Earth to the perihelion of Mars, and for an excess of 0·35 km./sec. a similar missile could reach the aphelion of Mars. These figures assume a take-off from Earth at the mean distance of this planet from the Sun.

If the missile was projected at the correct time it could be made to rendezvous with Mars and hence could possibly obtain close details of the planetary surface by means of an optical scanning device which would send the data back to Earth by using a system similar to that for the facsimile transmission of news pictures.[29]

An analysis of the size of the Martian probe vehicle shows that for a payload of one unit a take-off weight of 1,275 units would be needed to place the payload into a satellite orbit around Mars. It has also been suggested that the vehicle need not necessarily remain circling Mars but could, after a close approach to the planet, be allowed to follow its elliptical orbit and return to the vicinity of the Earth. After a period of two probe revolutions and three Earth revolutions the probe would approach close to this planet and the data could then be transferred. This would relieve some of the difficulties of generating sufficient power to telemeter the data from the distance of Mars. Indeed television pictures might be taken and stored on magnetic tape until the probe was close enough to Earth for them to be transmitted.

Yet other missiles will be sent to plunge into the atmospheres of Venus and Jupiter and at an even later stage one may be despatched to explore the outer atmosphere of the Sun.

Although chemical rockets would appear severely to limit man's interplanetary aspirations[21] it is found that the same rockets, if they carry instruments on one-way missions, can make it possible for data to be obtained concerning the planets of the solar system. Moreover, they will be able to do this long before true spaceships can be evolved. The serious difficulty in the way of this development is that of providing accurate control over interplanetary distances. Especially must this be borne in mind when it is realized that high-altitude experiments have shown the simple instrumentations to be the most reliable. Simplicity has been found to pay dividends in producing reliable data. It is, therefore, now a problem for the physicists to devise simple experiments which might be undertaken by probe missiles to obtain valuable planetary and lunar data which are never likely to be obtained from observations made at the surface of the Earth. The present-day limitations in rocket technology make it appear that probe missiles rather than spaceships will have to satisfy man's interplanetary longings at least for several decades until radically new forms of propulsion are developed.

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TECHNICAL REVIEW

Compiled by E. T. B. SMITH, B.Sc., A.F.R.Ae.S., G.I.Mech.E.

Satellite Vehicles

We have yet another batch of information on the Vanguard project, this time from speakers at a symposium organized by the (U.S.) Institute of Radio Engineers, as reported in Aviation Week of March 26, 1956. Milton W. Rosen of the N.R.L. opened by comparing Vanguard with Bumper-Wac, and pointed out that while the latter got to 250 miles altitude with a maximum velocity of 9,000 ft./sec., Vanguard would have to get to 300 miles altitude and still have a velocity of greater than 25,000 ft./sec. He then went on to describe Vanguard in some detail, and pointed out that it is a three-stage finless vehicle with guidance of the first two stages; the guidance equipment being located in the second stage. There will be a system of inertial guidance which will act upon the gimbaled motors of the first two stages, by means of electro-hydraulic systems. Roll control will be applied by small tangential rockets, and these will also be used to impart spin to stabilize the third stage before it is fired. Before launching, the complete three-stage vehicle will be 72 ft. long and 45 in. diameter at its maximum point.

As the stages burn out and drop off, the vehicle increases in velocity, and at the first stage burn-out 15 per cent. of orbital velocity is achieved; at the end of the second stage 32 per cent., while the remainder is provided by the third stage. After a vertical launch, from the Patrick Air Force Base in Florida, the vehicle will be tilted gradually to take up its orbit. After separation of the first stage 36 miles up at an inclination of about 45°, it will fall to earth (or into the Caribbean sea) about 230 miles from launch: the second stage flattens the trajectory and burns out at 140 miles altitude, and then coasts to 300 miles high and 700 miles further along, at which point the third stage will fire. The accumulation of errors in orientation of the second stage, the determination of the zenith of the trajectory of the second stage, and the stability of the third stage during burning, will make the exact positioning of the satellite at a given altitude quite impossible, so the intention is to fire the vehicle with a maximum velocity greater than that needed for a purely circular orbit at 300 miles altitude; the orbit will, in fact, be an ellipse with a perigee of not less than 200 miles altitude and an apogee of up to 1,400 miles altitude.

John P. Hagen of N.R.L. said that if the perigee is 300 miles then the life of the satellite will be about one year, but that if it is 200 miles up it will last for only 15 days, and at 100 miles perigee it will stay there for less than one hour.

Dr. John T. Mengel of the Naval Research Laboratory discussed the methods of tracking the satellite. The main system for establishing the position and orbit will be a device called Minitrack, which is a phase comparison radio triangulation technique. It requires a system of elaborate antennae at each ground station but the satellite transmitter is very light. The system works on 108 megacycles, and the continuous-wave satellite transmitter will have an output of between 10 and 15 milliwatts; either sub-miniature thermionic valves or transistors will be used, and with a total weight of 2 and 3 pounds the system will operate for several weeks. Each ground station will have two pairs of antennae at right angles to one another to determine the satellite's position in the north-south and east-west directions, relative to the zenith, by comparing the relative phase of the satellite signals received at each antennae. Another set of three antennae will be required at each station to resolve the ambiguities inherent in such phase comparison systems. Several dozen stations will be installed along the 75th meridian. Each will be linked to a central computing laboratory where the orbit and position calculations will be made.

As well as these Minitrack stations, it is hoped that radio amateurs will be able to provide data by simple observations with two-antenna systems. Optical systems will also be used for tracking, but it is expected that Minitrack will be able to establish the deviation of the Earth's surface from sphericity to an accuracy never before achieved. Satellite positions should be capable of establishment within a fraction of a milliradian. While Minitrack will be able to do a small amount of telemetering, the major amount of data will be transmitted by an FM pulse-width system on about 220 megacycles.

Tentatively, the following research groups have been allocated instrumentation space in one or more satellites:—U.S. Army Signal Corps Laboratory—percentage of earth covered by clouds; U.S. Air Force Cambridge Research Center—presence and size of interplanetary dust, and extreme ultra-violet radiation; Ballistic Research Laboratory, Aberdeen Proving Grounds—electron density above the F-layer region; University of Maryland—meteoric erosion of the satellite; N.R.L.—intensity of solar and cosmic ray radiation, upper atmosphere heating effects on satellite, and upper atmosphere pressure; State University of Iowa—composition of primary cosmic radiation.

Work with Aerobee

Aerobee-Hi has been stated in Washington to be the largest rocket that the United States will use during the IGY upper atmosphere tests. This rocket is about 37 ft. long and 15 in. diameter, and, together with a three-second solid propellant booster, is propelled by a motor burning RFNA and an aniline/alcohol mixture which gives about 4,000 lb. thrust for 53 seconds. Helium is used to inject the propellants.

An Aerobee launched from Holloman Air Force Base has been used to carry 18 lb. of nitric oxide gas to an altitude of 60 miles. On release, the gas acted as a catalyst to permit the atomic oxygen in the atmosphere, produced by

ultra-violet bombardment, to recombine into molecules. The energy so released by the oxygen produced a bright light which developed into a ball of fire about three miles in diameter before it faded. The local gas temperature (if temperature is a meaningful concept at such low pressures) is stated to have been "several hundred thousand degrees centigrade." It is possible that energy released in this manner might find some use in upper atmosphere propulsion.

Earth Satellite Issue of "Jet Propulsion," May, 1956

This issue, unfortunately not as readily available in this country as it might be, contains several interesting papers on earth satellites and their performance. In very brief summary, the following notes on papers may be made:—

N. V. Petersen—*Lifetimes of Satellites in near-circular and elliptic orbits.*

Density of upper atmosphere assumed is that of Grimminger (RAND Report R-105): shown that spherical bodies in circular orbits have lifetimes of about 25 days for Vanguard (22 lb., 21 in. diameter) at 200 miles. Increasing height and mass per unit area increase the lifetime, until with 320 lb./sq. ft. frontal area at 400 miles the lifetime approaches 30 years. Elliptical orbits give lifetimes affected most by perigee altitude: a sphere of 40 lb./sq. ft. falls at 66 miles per day from a perigee of 100 miles, but only at 50 miles per *year* from a perigee of 200 miles, the apogee in each case being 1,000 miles. Data are also considered for Ehricke's satelloid and Singer's MOUSE.

F. M. Perkins—*Flight Mechanics of Ascending Satellite Vehicles.*

Describes types of orbits and ascending trajectories and methods of computing them by analogue and digital methods, including the optimization of a vacuum stage thrust pitch programme for continuous thrust up to the orbit, and of the trajectory to minimize propellant consumption for a partly-coasting vehicle.

Jorgen Jensen—*Satellite Ascent Mechanics.*

Accuracy of guidance (altitude, velocity and direction) needed to inject a satellite into an orbit, given tolerances on the apogee and perigee altitudes and on the azimuth angle of the orbit; shown that with orbital injection point at 300 nautical miles altitude, perigee limit of 200 nautical miles and an apogee limit of 800 nautical miles, tolerances are needed on velocity and elevation of ± 1 per cent. and $\pm 2\frac{1}{4}°$ to maintain the satellite within these limits. Errors affect orbit shape rather than period of revolution.

E. Stuhlinger—*Control and Power Supply Problems of Instrumented Satellites.*

Compares photoelectric, solar/thermocouple and isotope/thermocouple systems, concludes that in each case overall efficiency is about 5 per cent. while weight is between $1\frac{1}{2}$ and $2\frac{1}{2}$ lb./watt (based on power plant for 100 watts).

The group of papers concludes with an extensive bibliography "History of the Artificial Satellite" by A. R. Krull, which covers the period 1879 to 1955 and appears to be a valuable compilation.

An Alternative Method of Rocket Staging

Cole and Epstein in a Technical Note in *Jet Propulsion* for March, 1956, discuss a method of rocket staging alternative to the conventional tandem arrangement. It is proposed that a number of identical stages to be disposed laterally, and ignited all simultaneously. Each stage is provided with propellant tanks, and as soon as firing starts propellants are pumped from the outer stages to the inner stages, the flow rate being adjusted so that the inner tanks remain full. As soon as the propellants in the outer stages are exhausted (the rocket arrangement being assumed laterally symmetrical about the central, last, stage), these stages are jettisoned, and this is repeated for subsequent stages. It appears that such a configuration may give an improved all-burnt velocity, as compared with the tandem layout, but it is the reviewer's opinion that the added complexity of transfer pumps and self-separating couplings would increase the structure weight, while the difficulty of obtaining good thrust balance between the widely separated members of a pair of motors constituting a stage would make the control of trajectory very difficult.

Moon Rockets

George H. Clement of RAND Corporation told the Franklin Institute at a symposium (*AW*, April 23, 1956) that the state of the art is sufficiently advanced for design work to begin on a rocket to hit the moon. He indicated that a 100 lb. payload split half and half between scientific instruments and a 1 watt telemetry set could be safely landed on the moon, using a three-stage rocket which would weigh 950,000 lb. at take off and reach 35,000 ft./sec. velocity at 350 miles above the Earth's surface. The third stage would be spin-stabilized and would carry an arresting rocket to lower the payload on to the Moon's surface: the trip would take about 56 hours, and extreme accuracy is needed in control of initial trajectory and speed. If the velocity at the 350 mile altitude reference point were 85 ft./sec. too great, the payload would miss the Moon and come back to the Earth.

Liquid Propellant Progress

Some improvements and developments in liquid propellants may be noted (*J.P.*, May, 1956). The Armour Research Foundation in Chicago has made 100 per cent. pure liquid ozone (O_3): this is apparently stable if the impurities can be kept below 20 p.p.m. total. Mixture of ozone and oxygen (38 per cent. by volume ozone) detonate with very light spark ignition, producing wave velocities at 24,000 cm./sec. (790 ft./sec.) and of up to 215,000 cm./sec. (7,000 ft./sec.) in violent detonation. The Linde Air Products Co. have investigated liquid ozone, and have concluded that even the highest purity ozone occasionally detonates, and that no desensitizer has been found.

The Becco Chemical Division, Buffalo, N.Y., is producing 90–100 per cent. hydrogen peroxide, as compared with the normally available 85–90 per cent. concentration.

Re-entry into the Atmosphere

In a Technical Note (*J.P.*, May, 1956) Dean, Reinhardt and Stehlinger present some results and calculations on the problem of preventing the volatilization of a satellite on re-entry into the atmosphere by frictional heating. They conclude that, for vertical entry at 100 miles altitude and velocity 4 miles/sec., a 100 lb. vehicle would have to absorb about 86 Btu/sq. in./sec., an equal amount of heat being given to the air. By allowing the sublimation of a beryllia nose cone (beryllia, on sublimation, absorbs 1,600 Btu/lb. at 4,500° F.), a mass of this substance of 46·5 lb. would absorb all the generated heat. This calculation assumed that the vehicle took 40 seconds to complete its downward path, and it is pointed out that a shallow spiral path with a time of 400 seconds would reduce the heat flux to 8·6 Btu/sq. in./sec., which is about the level encountered in some rocket motors. Difficulties would arise due to aerodynamic loading and thermal shock, and the proportionate weight of the coating seems very high.

Rocket Performance at Very Low Atmospheric Pressures

With most propellants, the difficulty of securing successful ignition is increased as the ambient air pressure in the combustion chamber is reduced, since it becomes more difficult to get intimate mixing of fair quantities of propellants and most thermal ignition systems do not work well. In order to do some necessary tests, the Rocketdyne Division of North American Aviation have added a facility at their Santa Susana plant which consists of a very large vacuum chamber (20 ft. long by 8 ft. diameter) into which the nozzle end of ICBM motors will be inserted. High capacity vacuum pumps will be used to produce very low pressures for starting, and presumably the chamber will be arranged so that free access to the atmosphere is given as soon as combustion is properly started. It is believed that as soon as this equipment is working, a more ambitious structure will be designed in which a complete motor run can be made under low pressure conditions (*AW*, April 23, 1956).

Electric Batteries using Radioactivity

A report in *Chemical Engineering Progress* for February, 1956, states that nuclear sources will take some considerable time before they are available for use in electric batteries. It appears that, of all types considered (direct charging with charged particles, contact potentials, thermocouples, and semiconductor junctions), only the last two may have some utility although the efficiency is of the order of 1 per cent. (electric power as fraction of heat release) and there are other difficulties concerned with cost, shielding, and radiation damage to the materials of the battery. However, the semiconductor junction battery can give enough power to run transistorized radio equipment.

German High Altitude Rocket Firings

The German Association of Rocket Engineers (AFRA) plans to fire a three-stage rocket which will reach a height of at least 60,000 ft. during the International Geophysical Year. Firings will be made from the base near Cuxhaven (*AW*, March 26, 1956).

ABSTRACTS

Edited by J. HUMPHRIES

AIRCRAFT

(258) **The prospects and problems of rocket propulsion for aircraft.** A. D. Baxter. *J. Roy. Aero. Soc.*, **59**, 315–332 (*May*, 1955). Compares the performance of different aircraft propulsion systems and outlines the field of application of rocket motors. Take-off assistance, boost of climb and level flight acceleration are the obvious present-day applications, but under special conditions transport aircraft of large size would be economically powered by rockets. Describes different rocket units, and the advantages of composite units, ducted rockets, rocket-ramjets. Combustion chambers, injectors, and fuel supply arrangements are described, and means of motor control. The paper is followed by a discussion. (9 *refs.*)

(259) **Optimum climbing technique for a rocket-powered aircraft.** A. Miele. *Jet Propulsion*, **25**, 385–391, 399 (*Aug.*, 1955). Contains a preliminary investigation of the flight path which mimimises both the time and the propellant expenditure necessay to fly a rocket-powered aircraft from one given combination of speed and altitude to another. (17 *refs.*)

(260) **Power plants for supersonic flight.** E. S. Moult. *D.H. Gaz.* **88**, 85–92 (Aug., 1955). Discusses the power plants available for flight at up to M = 3 and the best combinations for specific purposes. Problems of ambient temperature and high-temperature materials are considered. Concludes that the turbojet with after-burning will meet most requirements for some time, but that the ultimate is the rocket motor used in conjunction with a turbojet engine.

ASTRONAUTICS

(261) **Optimal programming of rocket thrust direction.** D. F. Lawden. *Astronautica Acta*, **1**, 33–56 (1954). The paper discusses the best programme for rocket thrust direction if the rocket motor has a fixed thrust programme. It is deduced that it is advantageous to displace the line of thrust from the direction of motion by a small angle in the sense of the gravitational field and to diminish this angle to zero as the manoeuvre proceeds. Numerical examples are given. (9 *refs.*)

(262) **World astronauts meet in Denmark.** F. I. Ordway and H. E. Canney. *Sky and Telescope*, **15** (1), 14–15 (1955). Review of proceedings.

(263) **Some problems in space travel.** J. G. Porter. *J. Inst. Navig.*, **8**, 224–235 (*July*, 1955). Describes the conditions required for the successful achievement of a flight from one planet to another, and underlines the need for exceptional accuracy in determining the speed and angle of launch. Means for fixing the position and velocity of a spaceship are discussed.

(264) **The shot to be seen around the World.** M. Summerfield. *Jet Propulsion*, **25**, 369 (*Aug.*, 1955). A report, with comment, of the announcement of the U.S. Satellite vehicle programme in July, 1955.

(265) **Sixth International Astronautical Congress.** E. Burgess. *Aircraft Engng.*, **27**, 348–349 (*Dec.*, 1955). Abstracts of some of the technical papers presented at the congress.

(266) **International law and activities in space.** C. W. Jenks. *International and Company Law Quarterly*, 99–114 (*Jan.*, 1956).

(267) **Design criteria for minimum satellites.** S. F. Singer. *Aero Dig.*, **72** (4), 36–37 (*April*, 1956). Classifies vehicles on basis of weight, telemetering system and orientation requirements.

(268) **Relativity and space travel.** H. Dingle and W. M. McCrea. *Nature*, **177**, 782–785 (*April* 28, 1956). Correspondence regarding the difference in elapsed time experienced by observers at rest and at speeds approaching that of light. (7 *refs.*)

(269) **Next satellite problem: data descent.** R. Cushman. *Aviation Wk.*, **64** (20), 53, 56–57 (*May* 14, 1956). Method of data recovery proposed by Porter involves the use of an inflatable stainless-steel balloon. This would be retarded initially by a small rocket and then allowed to fall free through the atmosphere. Maximum stagnation temperature on a 3-ft. diameter balloon weighing 7 lb. would be 3,000° R.

ASTRONOMY

(270) **Are planets habitable ?** J. R. Randolph. *Ordnance*, **40**, 608-610 (*Jan.-Feb.*, 1956). Brief review of existing knowledge.

BIOLOGY AND MEDICINE

(271) **Cosmic rays. Physiopathological effects on man.** P. Gaspa. *Météorologie*, **4**, 261–271 (1953) (*In French*). Survey of the problems from the point of view of interplanetary flight.

(272) **Space equivalent conditions within the Earth's atmosphere, physiological aspects.** H. Strughold. *Astronautica Acta*, **1**, 32–40 (1954). Conditions in free space differ in many ways from those found at the bottom of the atmosphere, but certain of these differences become fully significant even within the limits of present-day flight, e.g. anoxia, boiling of body fluids, and the impossibility of using ambient air for cabin pressurization. The appearance of cosmic rays and meteors as altitude increases are other factors, and each of the foregoing constitutes a particular "static space-equivalent condition." The state of zero-gravity is a "dynamic space equivalent condition" and does not depend on height or distance from the Earth. (25 *refs.*).

CHEMISTRY

(273) **Explosion limits of ozone-oxygen mixture.** G. A. Cook, E. Spadinger, A. D. Kiffer and C. V. Klumpp. *Industr. Engng. Chem.*, **48**, 736–741 (*April*, 1956). Safety limits are given for various temperatures. Increased purity reduces the possibility of a chance explosion, but it is recommended that the pure liquid be handled behind suitable barriers. (23 refs.)

(274) **Thermal decomposition of simple nitrite esters.** J. B. Levy. *Industr. Engng. Chem.*, **48**, 762–765 (*April*, 1956). Decomposition products of *tert*-butyl, isopropyl and *n*-propyl nitrites are given and discussed. (7 *refs.*)

(275) **Preparation of liquid ozone and ozone-oxygen mixtures for rocket application.** G. M. Platz and C. K. Hersh. *Industr. Engng. Chem.*, **48**, 742–744 (*April*, 1956). The method for producing nonsensitized ozone consists of purification of the process oxygen by high-temperature decomposition of sensitizing organic materials and maintaining a system such as precludes any possible recontamination. The principal compounds which sensitize ozone are organic and must be maintained below a concentration of 20 p.p.m. expressed as carbon dioxide, to ensure production of stable ozone. (12 *refs.*)

(276) **Decomposition of carbon-nitro compounds.** K. A. Wilde. *Industr. Engng. Chem.*, **48**, 769–773 (*April*, 1956). The thermal decomposition of nitroethane, 1-nitropropane and 2-nitropropane, has been studied in a flow system. Results are presented and discussed. (11 *refs.*)

MATERIALS

(277) **Rupture of heat-resistant alloys in flame gas atmospheres.** E. W. Larocca. *Jet Propulsion*, **25**, 396–399 (*Aug.*, 1955). Reports an investigation into the failure of two cold-worked commercial alloys when stressed in a propane combustion zone. Rupture times between 30 seconds and 1 hour were obtained, and the results are in general agreement with conventional stress-rupture and creep data. (9 *refs.*)

PHYSICS

(278) **On the investigation of heavy primary cosmic rays by means of a colour method.** J. Eugster. *Astronautica Acta*, **1**, 57–60 (1954) (*In German*). Photographic plates have certain disadvantages when used to record cosmic-ray particles in biological research, particularly where there is a delay between exposure and development. This paper describes a new method based on the use of dyes to detect particle tracks, which may be applied more widely, even to events happening within living cells.

(279) **The investigation of the initiation of combustion processes.** I. Sänger-Bredt, *Austronautica Acta*, **1**, 3–31 (1954) (*In German*). An examination of experimental methods for the investigation of reactions between individual molecules, and the light they throw upon the the mechanism of energy release in chemical combustion processes. Also proposes the use of crossed molecular beams, as an extension of present methods. (152 *refs.*)

(280) **Infrared spectra of propellant flames.** A. D. Dickson, B. L. Crawford and D. L. Rotenberg. *Industr. Engng. Chem.*, **48**, 759–761 (*April*, 1956). Solid propellant strands were burned in a vessel that permitted simultaneous observation by a rapid-scan spectrometer and a movie camera. Results are presented. (3 *refs.*)

(281) **Nitrate ester flames.** R. Steinberger. *Industr. Engng. Chem.*, **48**, 766–768 (*April*, 1956). Review of current fundamental work. (20 *refs.*)

(282) **Towards flight without stress or strain—or weight.** *Interavia*, **11**, 373–4 (*May*, 1950). Reports that the weights of some materials have been cut by up to 30 per cent. by "energizing" them and suggests that "electrogravitics" may revolutionize methods of transport by control of gravity fields.

(283) **Study of solar radiation at high altitudes using rocket-borne instruments.** R. L. F. Boyd. *Proc. Roy. Soc.*, **236**, 211–216 (*Aug.* 2, 1956). Review of past work. (4 *refs.*)

PROJECTILES

(284) **Development of a stabilization system for the Viking rocket.** N. E. Felt. *Jet Propulsion*, **25**, 392–395, 399 (*Aug.*, 1955). The Viking is stablized in its near-vertical flight by a stabilization system involving control of motor thrust direction, roll control by aero-dynamic surfaces in the early part of flight, and a hydrogen peroxide jet control system for both roll and lateral stablization. The paper describes these systems and the design methods used.

(285) **Flight-control apparatus involving steering combustion chambers.** R. H. Goddard. *U.S. Pat. No.* 2,726,510 (*Dec.* 13, 1955). The use of four auxiliary chambers placed in the nose for steering a rocket vehicle. Propellant flow to the chambers is varied to produce a steering moment.

(286) **Rocket exploration of the upper atmosphere.** F. E. Jones and H. S. W. Massey. *Nature*, **177**, 643–645 (*April* 7, 1956). Description of the Gassiot high-altitude vehicle and its launching gear—performance curves are given.

(287) **Delta fin Veronique aims at 137 miles.** F. I. Ordway. *Aviation Wk.*, **64** (22), 56–57, 59, 61, 63 (*May* 28, 1956). Detailed description of French high-altitude research rocket with tabulated results of all firings to date.

(288) **The ballistic missiles.** M. S. Watson. *Ordnance*, **40**, 948–952 (*May-June*, 1956). General review of present position and potentialities.

(289) **Designing for high-speed missile cooling.** R. H. Hepper. *Aero Dig.*, **72** (6), 48–50, 52, 53 (*June*, 1956). The use of both insulation by means of nonstructural outer shell or insulating coating and cooling methods are discussed. Potential nose and leading edge materials are reviewed, these are mainly ceramics. Thermal stresses can be minimized by using monococque construction. (23 *refs.*)

(290) **Corporal gives Army nuclear capability.** D. A. Anderton. *Aviation Wk.*, **65** (2), 50–51, 53, 55, 57 (*July* 9, 1956). Full description of auxiliary equipment used with Corporal and mode of operation.

RADIO AND ELECTRONICS

(291) **Launching control for guided missiles.** J. B. Schrock. *Electronics*, **28** (2), 122–127 (*Feb.* 1955). Master timing circuits which actuate guidance and telemetering equipment and alert rocket-range control units. They also have fail-safe and interlocking provisions to guard against premature ignition.

(292) **Data reduction system for missile telemetering.** E. M. McCormick. *Electronics*, **28** (5) 126–130 (*May*, 1955). Flight data telemetered from experimental missiles is decommutated and recorded on punch-cards for further study by f.m. data converter and data processing equipment. Designed for f.m./f.m. telemetering, the system also handles pulse-width signals.

(293) **Antenna system for missile telemetering.** G. E. Bower and J. B. Wynn. *Electronics*, **28** (6), 164–167 (*June*, 1955). Dual helical antennas, low-noise preamplifier and multi-coupler for ground receiving station provide overall gain of over 30 db.

(294) **Modular simulator tests missile radar.** M. Krakauer and R. J. Bibbero. *Electronics*, **28** (7), 127–129 (*July*, 1955).

(295) **Shipboard telemetering for Terrier missiles.** W. S. Bell and G. W. Schultz. *Electronics*, **29** (6), 134–137 (*June*, 1956). An automatic f.m./f.m. telemetering system

serving both launchers on U.S.S. *Mississippi* provides magnetic tape recordings of internally derived data for fleet evaluation of production-type Terrier missiles without interfering with missile fire control system. Six channels accommodate six missiles simultaneously.

ROCKET MOTORS

See also abs. no. 306.

(296) **The de Havilland Super Sprite.** *D.H. Gaz.*, **88**, 93–95 (*Aug.*, 1955). A description of this assisted take-off rocket power unit. Propellants are hydrogen peroxide and kerosine, it produces 4,200 lb. thrust for 40 seconds, weighs 620 lb. dry. The starting sequence and recovery procedure are also described.

(297) **Chemical engineering in jet propulsion.** L. R. Rapp. *Chem. Engng. Progr.*, **52**, 143–148 (*April*, 1956). Deals mainly with the elementary design problems of rocket combustion chambers and choice of propellants. (2 *refs.*)

ROCKET PROPELLANTS

See also abs. nos. 275, 280 and 297.

(298) **Ignition delay of nonhypergolic systems.** R. Grollier-Baron and G. Wessels. *Mém. Poudr.*, **36**, 285–304 (1954) (*In French*). Nitric acid and gas-oil were injected through impinging jets into a heated chamber and ignition delays determined by measuring the distance between the points of mixing and ignition.

(299) **Small-scale facilities for development of double-base propellants.** A. T. Camp, C. H. Carlton, Q. Elliott and J. H. Wiegand. *Chem. Engng. Progr.*, **52** (2), 79F–82F (*Feb.*, 1956). Properties of double-base propellants are discussed and plant for making up to 10 lb. batches is described. By use of the strand-burning test it is possible to go directly from small-scale to full-scale manufacture with complete confidence. (3 *refs.*)

(300) **Fuel-binder requirements for composite propellants.** W. F. Arendale. *Industr. Engng. Chem.*, **48**, 725–726 (*April*, 1956). The polymer used as a binder should have the properties of an elastomer with good low temperature properties, resistance to flow at high temperatures, high fuel value, and must be adaptable to the manufacture of the design required for thrust units.

(301) **Rheology of composite solid propellants.** P. J. Blatz. *Industr. Engng. Chem.*, **48**, 727–729 (*April*, 1956). A tool used in the study, the method of reducing the data, and the parameters which chararacterize some typical composite propellants are described. (3 *refs.*)

(302) **Radiographic inspection of loaded rocket motors.** J. Buchanan and B. D. Herbert. *Industr. Engng. Chem.*, **48**, 730–731 (*April*, 1956). Solid propellant motors using a propellant grain formed directly in the case are successfully inspected for critical voids and irregularities by modern radiographic equipment. Gamma ray sources are used for inspection of large motors.

(303) **Concentrated hydrogen peroxide as a propellant.** N. S. Davis & J. H. Keefe. *Industr. Engng. Chem.*, **48**, 745–748 (*April*, 1956). Hydrogen peroxide (90 per cent.) has been used as a source of auxiliary power for helicopters, for propelling models, for submarine propulsion, and in rocket-assisted take-off units. Its physical properties and performance, both as a mono-propellant and in conjunction with various fuels are discussed. (5 *refs.*)

(304) **Performance characteristics and limitations of chemical propellants.** A. J. Stosick. *Industr. Engng. Chem.*, **48**, 722–724 (*April*, 1956). The thermodynamic requirements are reviewed and it is stressed that ultimate limits are nearly approached by present systems. Combustion characteristics of liquid propellants depend primarily on heat and mass transfer processes, rather than on chemical details. Logistic considerations are of dominant importance in selecting liquid propellants. Methods of improving the performance of solid propellants are discussed.

(305) **Liquid propellant handling, transfer and storage**. P. M. Terlizzi and H. Streim. *Industr. Engng. Chem.*, **48**, 774–777 (*April*, 1956). Discusses quality control, transfer operations, personnel protection, storage, disposal and decontamination, fire-fighting and materials of construction.

(306) **Investigation of combustion in rocket thrust chambers.** C. H. Trent. *Industr. Engng. Chem.*, **48**, 749–758 (*April*, 1956). The results of an experimental investigation of the high pressure combustion of white fuming nitric acids and hydrocarbons is presented.

The utilization of a water-cooled probe allows the measurement during steady state operation of the temperature, pressure, gas velocity and chemical composition of the gases. Temperature and velocity profiles are presented and an attempt is made to relate kinetic information to combustion chamber design. The relationship between the injector and the combustion process is discussed. Results indicate that heterogeneous combustion occurs for approximately the first 50 per cent. of propellant conversion. The design of the injector has little effect on the gas phase chemistry of the combustion process. (14 *refs.*)

(307) **Tetranitromethane as oxidizer in rocket propellants.** J. G. Tschinkel. *Industr. Engng. Chem.*, **48**, 732–734 (*April*, 1956). The performance of various fuels with both liquid oxygen and tetranitromethane as oxidants are compared. Using a V-2 size vehicle it is shown that tetranitromethane gives a 22 per cent. greater range than liquid oxygen. However, it is not in large-scale production, has a high freezing point and tends to be shock sensitive. (15 *refs.*)

REVIEWS

The Report on Unidentified Flying Objects

(By Edward J. Ruppelt. 1956. 315 pp. Victor Gollancz, London. 18s. 6d.)

It is a great pity that this book was not published several years ago before the "flying saucer" situation got out of hand and the influx of hoaxers, credulous wishful-thinkers and plain lunatics made it impossible to take the subject seriously. Captain Ruppelt was formerly head of the American Air Force Project Blue Book, set up to investigate reports of U.F.O's, and is therefore in a unique position to write on the subject. Indeed, one is tempted to say that he is the *only* author of a book on U.F.O.'s in a position to write about them, and it would clear the air (in more ways than one) if all earlier volumes could be mercifully forgotten.

The fascinating thing about Captain Ruppelt's book is that after spending several years and a lot of Air Force money on the job he still doesn't know what U.F.O.'s are. Some of the completely authenticated sightings he reports (several of which involved simultaneous visual and radar observations) are quite inexplicable. Unless there is an improbable amount of synchronized and motiveless lying in progress, the fact must be squarely faced that there are some very odd things going on in our atmosphere.

One of the most valuable aspects of this book is the background it gives to the entire U.F.O. investigation. The number of experts and research organizations that have become entangled at one time or another is considerable. (Some are not named, and readers may have fun identifying them by their context.) Even the massed brains of the RAND Corporation, for example, were unable to do anything about the 23 per cent. of unexplainable U.F.O.'s.

Much of Captain Ruppelt's book is not very complimentary to the U.S. intelligence services, but it must be admitted that they were faced with a fantastic and unprecedented situation. The Air Force did not help matters by issuing ridiculous explanations of U.F.O.'s, weasel-worded statements, and in general by changing the party-line. For example, by the end of 1948 Air Force Intelligence had submitted a Top Secret report concluding that U.F.O.'s *were* of interplanetary origin, but was unable to convince General Vandenburg, then Chief of Staff. Thereafter the whole subject became so involved in

Pentagon politics that the people who believed in U.F.O.'s were scarcely on speaking terms with those who thought the whole business a lot of nonsense.

Of particular interest is the result of a poll of 45 leading American astronomers. 11 per cent. had seen U.F.O.'s themselves, and 23 per cent. thought that they were a much more serious problem than most people recognized. None, however, considered that they were spaceships.

Captain Ruppelt does a very creditable job of fence-sitting, but it is obvious that he will not be at all surprised if U.F.O.'s do turn out to be spaceships. We wish that his fascinating and frequently amusing book could be the last word on the subject until more evidence accumulates, but this is rather too much to hope. Meanwhile let us all look forward with as much patience as we can to the day when U.F.O.'s cease to be U and become Non-U. When the satellite observing teams start work around the end of 1957 the solution of this really baffling problem may be in sight.

A. C. CLARKE.

The Exploration of Mars

(By Willy Ley and Wernher von Braun. With 16 colour and 5 black-and-white illustrations by Chesley Bonsteall. x + 176 pp., 50 plates. Sidgwick & Jackson, Ltd., London, 1956. 30s.)

In the reviewer's opinion, this is the best of the books in this series since the incomparable *Conquest of Space*. Possibly this may be due to the fact that there are fewer cooks making the broth; despite the wide ground it covers, the volume seems more homogeneous and well-integrated than its last two predecessors.

The first half of the book is devoted to a thorough discussion of Mars, describing in detail its movements, the history of Martian observation, the famous canal controversy and the arguments for and against the existence of life on the planet. Though there is little new that can be said on the subject, the survey is both thorough and entertaining, and contains many of those snippets of historical erudition we have come to expect from Willy Ley. There is also a very comprehensive bibliography of all books of any importance on the subject of Mars from 1737 onwards.

Mr. Ley is wise enough not to commit himself on the subject of the canals; incidentally, he gives the Italian meaning of the unfortunate word *canali* as "grooves." How the history of Martian observation might have been changed had Schiaparelli chosen a word which aroused fewer echoes in Anglo-American minds! (Yet perhaps it was a fortunate accident after all. There would never have been so much interest in Mars—or in space-travel—without that provocative label.)

The second half of the book is concerned with Dr. von Braun's proposals for a chemically-fuelled expedition to Mars, and his ideas have changed considerably from those put forward in *The Mars Project* (1953), being now much more realistic—though still ambitious enough, since they involve a total expenditure of half a million tons of propellant! This, however, is only a tenth of the quantity needed in *Mars Project*, and the enormous saving has been

achieved by using two ships carrying a dozen men, instead of ten ships carrying seventy. The engineering assumptions, including the choice of propellants (nitric acid and hydrazine) remain unchanged.

Both ships envisaged are of the open-structured "deep-space" type and are assembled in an orbit at a height of 1,075 miles. One is a passenger-carrying vessel which will make the round trip; the other, the "cargo ship," carries the stores needed for the journey and also the landing-craft which will make the descent to Mars. The ships weigh 1,870 tons each on departure from the Earth orbit, and empty propellant tanks and surplus motors are jettisoned at the earliest possible moment.

The most impressive item of the expedition is the 177-ton glider which will descend to Mars when an orbit has been established around the planet at a height of 620 miles. It will carry nine of the expedition's twelve men, who will spend 400 days on Mars (a period determined by the length of time it takes for Earth and Mars to get into the right positions for the return journey).

The landing party will need 18·7 tons of oxygen, water and food, and 35 tons of equipment—including two small tractors to allow a good radius of action. This explains the size of the glider, which also has to carry sufficient propellant to get the nine men and 5·5 tons of specimens back to the 620 mile orbit. When it is time to take off from Mars, the glider is stripped down and winched into the vertical position, becoming in effect a simple rocket, which will be left orbiting Mars for the benefit of future expeditions.

The passenger ship then returns to Earth, but to economize propellants settles into an orbit at the height of 56,000 miles, from which crew and specimens are retrieved by a ferry-rocket launched from the lower 1,075-mile orbit.

This brief summary gives little idea of the detailed and careful thought that has obviously gone into the entire scheme. Five pages of closely-printed type give the weight-breakdown and performance data of the two ships, as well as of the orbital supply ships (weighing 1,410 tons) which have to be launched from Earth in the first place to establish the bridgehead in space.

It would not be hard to challenge the project on many minor points (how can one be sure at this early date that, for instance, 12 spacesuits will weigh 1·32 tons, or that the pressurized caterpillar tractors needed for the expedition will weigh only 3·3 tons each?) but that does not detract from the interest and merit of the conception as a whole. Dr. von Braun has made a brilliant attempt to show that, even with existing propellants and techniques, an expedition to Mars is within the bounds of engineering possibility. Since such a good case can be made to-day, one can have little doubt that after another decade or two of rocket development—with or without the arrival of nuclear propulsion—we will be in a position to do something serious about the tantalizing little red planet.

Chesley Bonestell's paintings, as ever, are both convincing and beautiful. One of them—a double-page spread showing the Martian landing site—must be among the finest he has ever done. It is also interesting to note several contributions from the ubiquitous Walt Disney, whose studios have provided photographs of some excellent models.

There are not many books on astronautics to-day which have something

new to say, or can put it as well as *The Exploration of Mars* does. It is strongly recommended to everyone interested in space-flight—particularly to any pessimist who is afraid we may get stuck at the Moon and won't be able to get any further afield.

A. C. CLARKE.

Solid Propellant Rockets

(By Alfred J. Zaehringer. Published by American Rocket Co. 162 pages. $4.0. 1955)

This book is very patchy in quality. It is on the whole well illustrated and some of the tabulated information on rocket motors and facilities is quite useful. There are, however, plenty of errors, e.g. in the section devoted to British developments on page 25, and the sections devoted to theory show that the author is not at home with his subject. Thus H in equation 2–8 is not "the actual heat energy released by a unit mass of propellant," even though it may approximate to it, and the calculations in Examples 1 and 2, on page 68 are incorrect; in Example 3 the choice of a reaction which gives a mainly solid product is rather unhappy in an illustration of the calculation of maximum theoretical exhaust velocity.

There is little in the book on motor design or internal ballistics although a good deal of information on these subjects has already been published in the open literature. On the other hand a considerable amount of space is devoted to extraneous topics such as guidance and control or to topics like rocket-motor instrumentation which, although important to rocket-motor development, are of an ancillary character. The author does something to compensate for omissions by the provision of a fairly extensive bibliography and the promise of two further books from his own pen.

To sum up it can be said that though it is claimed in the preface that the book is written for the engineer or scientist it is doubtful if it is of much value to this class of reader. On the other hand, the intelligent layman may find in it much of interest and although the style is sometimes a little peculiar the book is easy to read. The price seems rather high for a small paper-covered book.

DR. W. R. MAXWELL.

CORRESPONDENCE

A Famous Russian Encyclopaedia of Astronautics

SIR,

I was most interested to see the further article (*J.B.I.S.* **15** (2), 82) on the Rynin books. It would seem that in some respects the B.I.S. is more fortunate than we in the volumes they have, but we certainly have a copy of the first. Should any of your members wish to see it we are, of course, happy to make it available to them.

F. H. SOUTH,
(*Royal Aeronautical Society*).

4, Hamilton Place,
London, W.1.

Electrostatic Generator

SIR,

With regard to your recent correspondence on vacuum insulated rotating electrostatic generators, I should like to bring your attention to the fact that such a machine has been constructed in the High Voltage Lab. of Prof. John G. Trump at the Massachusetts Institute of Technology.

This machine is described, with illustrations, by Dr. Trump in a paper entitled "*Electrostatic Sources of Electric Power*" presented on January 29, 1947 at the Conference on energy sources held by the American I.E.E. during their winter meeting in New York City.

This synchronous generator delivered 60 watts at 90 kV., 60 cycles. Its voltage gradient was about 100 kV. per cm., and its efficiency over 99 per cent.

J. WATSON.

DR. BROSAN REPLIES:

As far as I know there is nothing new about the idea of electrostatic generators. However it is just not practical for the generation of substantial powers under terrestrial conditions because of the size of the vacuum enclosure.

An Analogue Computer

SIR,

May I congratulate Mr. Cross on his patience and ingenuity in the design and construction of the analogue computer described in his recent article (*J.B.I.S.* **15** (1),7). May I point out that the description of the computer as a "Link trainer" type of apparatus is hardly fair, since the "Link trainer" has now been developed into quite a complex and realistic affair. Its mechanism incidentally is mostly electrical because such equipment is at the same time more compact, and more flexible than other varieties of analogue. Incidentally electrons are cheap and plentiful, and do not require putting back at the end of each flight.

There are several points which I should like to make in connection with Mr. Cross' article.

Firstly it is a little dangerous to say that digital methods can never provide the opportunity of practice in a realistic manner. The only reason against the construction of digital simulators is that digital computers are not yet fast enough to perform the required calculations on a real time scale, but on a simple set-up such as Mr. Cross has, this may indeed be possible with the latest techniques.

Secondly, the introduction of an inverse square law for the gravitational acceleration would invalidate the integration of equation 1 to obtain equation 2, of which Mr. Cross' equipment is the analogue. Any equipment designed to cope with a varying gravitational acceleration would have to be an analogue of equation 1, and the presentation of altitude would then require two integrations in the equipment. This might prove tricky in an hydraulic analogue, although it presents no difficulties in an electrical system.

As regards the simulation of "g" by mounting the trainer in a centrifuge, one disadvantage of this system is that one cannot reduce the effective "g" below 1 by this means, other than transiently. This, I suggest, may be a serious drawback to a trainer for the purpose envisaged here. The only method I can see of overcoming it is to mount the whole thing, centrifuge, trainer, and trainee, in a space station where zero "g" conditions prevail.

Finally may I suggest that, not only will pilots be trained initially on a full scale rocket simulator, but that such a simulator may well be used to test prototype control and instrument systems before the first piloted rocket is designed. The possibility of obtaining pilot's comments on the use of such systems under fairly realistic conditions while still in the design stage, will prove invaluable to the engineers concerned. Already facilities exist for testing, e.g., guided weapon control systems by attaching them to an appropriately designed analogue computer, and it seems likely that this practice will spread.

Aylesbury, Bucks. C. D. ALLEN.

MR. CROSS REPLIES,—

I must disagree with Mr. Allen on one point. I think it *is* quite fair to describe this computer as a "Link Trainer" type of apparatus, for it is just that. Admittedly it is a much simpler and less realistic affair, but that is beside the point.

I am in agreement with the other points that Mr. Allen makes, but when he says that electricity is cheap that is not quite relevant. The apparatus for handling and measuring the flow of electrons is definitely expensive, and I do not believe that one could build the electrical equivalent of this computer for an outlay on materials of only £4 10s. 0d. However, I unreservedly endorse the view that the electrical machine is likely to be superior in most other respects.

I am greatly impressed by Mr. Allen's final point, particularly with the idea that automatic control systems as well as human pilots could be tested by these computer techniques. My experience of automatic control of chemical processes is that extensive preliminary adjustments are needed before such things work properly. It might well be a slow and expensive business learning the correct control settings by observing the misbehaviour of a series of trial rockets. I am very grateful to Mr. Allen for making these valid and interesting comments.

The following minor corrections have also been drawn to my attention:

(*a*) On page 9, equation (5) should read:
$X_0 = \frac{1}{2}gt^2 + ct$ etc, instead of $X_0 + \frac{1}{2}gt^2 + ct$ etc.

(*b*) On page 12, equation 10 and its preceeding line:

"Which gives on *integration*" should read: "Which gives on *differentiation*."

u is not explicitely defined in the paper, it is in fact the exhaust velocity of the rocket expressed as a hydraulic head.

Some Thoughts on Getting Back

SIR,

It is customary to carry out paper studies of possible space projects on the assumption that they would be successful. The purpose of this note is to draw attention to the need for considering the possibilities of failure and to suggest that this will influence the planning of a real project to a significant degree.

It has long been recognized that meteors present a hazard to space-flight, and the problem has received much attention. Less generally appreciated, perhaps, is the fact that engineering systems of the type employed in the aircraft and missile field are never 100 per cent. reliable. The consequences of a failure of a component in space may well be considerably more serious than the consequences of a failure in the air.

An interesting paper bearing on this general problem was prepared by the Aeronautical Engineering Department at M.I.T. and presented by Prof. P. E. Sandorff.[1] It contains the suggestion that the present scheme of launching rocket research aircraft from aircraft powered by air-breathing engines should be applied also to the launching of satellite rocket vehicles. Sandorff estimates that the probability of success for a 3-stage rocket designed to place 500 lb. in a circular orbit around the Earth is about 35 per cent., whereas that for a corresponding 2-stage rocket with aeroplane step is about 60 per cent.

If these views are accepted, then clearly the aeroplane step will continue to find favour, particularly for launching manned vehicles. The probability of failure is still appreciable, however, and the need to plan for recovery of the vehicle and crew is essential. An article in *Collier's*[2] dealt with the problem of the escape of crew from high-altitude craft, in capsules designed to provide the crew members with an artificial environment as in a spaceship cabin. Recovery of the vehicle should be attempted where possible. In this connection it seems that the final stage should be capable of landing on water and floating, as launching over the sea would give it a wide choice of landing area. During the latter stages of its descending glide, ramjet power could be usefully employed to provide additional range. This latter feature could, in any event, probably be incorporated with advantage in any vehicle intended to return to the Earth from space to permit compensation for deviations from a programmed flight path.

For long range, e.g. interplanetary, excursions into space, attention must be given to the need for adequate spares. One rather drastic, but possibly necessary, solution to the problem would be to venture forth with several ships with readily interchangeable components and indulge in "robbing" where necessary. In order to give the crew the best chance of returning when jettisoning becomes necessary, all gear not essential to the crew on journey should be stored in separate and readily jettisonable compartments.

It appears likely that considerations of the possible need to cancel a mission once it is under way may affect the choice of orbit. As an example of this, for very distant missions it is known that some economy in fuel may be effected by first travelling on a semi-ellipse which carries the vehicle close to the Sun before thrusting at perihelion into a new orbit. The selection of a semi-ellipse of three

months' duration would have the advantage of presenting the alternative of a return to Earth one year after departure by continuing in the elliptical orbit for two complete elliptical paths around the Sun if the decision to do so were made within three months of departure.

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